SPACE TRANSFER VEHICLE CONCEPTS AND REQUIREMENTS
NAS8-37856

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FOREWORD

This report, prepared by Martin Marietta Corporation, is submitted to George C. Marshall Space Flight Center, National Aeronautics and Space Administration (NASA), Marshall Space Flight Center (MSFC), Alabama, in response to the DR-5 requirements of contract NAS8-37856, Space Transfer Vehicle Concept and Requirements. It is the DR-5 identified in Data Procurement Document No. 709.

GLOSSARY

ACC	Aft Cargo Carrier
ACS	Attitude Control System
AFE	Aeroassist Flight Experiment
Al-Li	Aluminum Lithium
ALS	Advanced Launch System
APCM	Advanced Programs Cost Model
ASE	Airborne Support Equipment
ATLO	Acceptance, Test, Launch, and Operations
ATR	Advanced Technology Roadmap
BOE	Basis of Estimate
C&DM	Configuration and Data Management
CAD	Computer-Aided Design
CDR	Critical Design Review
CER	Cost Estimating Relationship
CG	Center of Gravity
CLAAS	Closed-Loop AeroAssist Simulation
CNDB	Civil Needs Data Base
COLD-SAT	Cryogenic Onorbit Liquid Depot Storage, Acquisition, and Transfer Satellite
CSLI	Civil Space Leadership Initiatives
DDT&E	Design, Development, Test, and Evaluation
DOD	Department of Defense
DR	Data Requirement
DRM	Design Reference Missions
ETO	Earth-to-Orbit
ETR	Eastern Test Range
GEO	Geosynchronous Earth Orbit
GN&C	Guidance, Navigation, and Control
GPS	Global Positioning Satellite
GSE	Ground Support Equipment
H/W	Hardware
I/F	Interface(s)
ILC	Initial Launch Capability
IMU	Inertial Measurement Unit
IR	Interim Review
IR&D	Independent Research and Development
IRD	Interface Requirements Document
KSC	Kennedy Space Center
L/D	Lift-to-Drag Ratio
LAD	Liquid Acquisition Devices
LCC	Life-Cycle Cost
LEO	Low-Earth Orbit

LeRC	Lewis Research Center (NASA)
LEV	Lunar Excursion Vehicle
LTV	Lunar Transfer Vehicle
LV	Launch Vehicle
MAP	Manifesting Assessment Program
MDC	McDonnell Douglas Corporation
MLI	Multilayer Insulation
MMS	Multimission Modular Spacecraft
MSFC	Marshall Space Flight Center
MSS	Manned Space Systems
NASA	National Aeronautics and Space Administration
NASP	National Aero-Space Plane
OTV	Orbital Transfer Vehicle
P/A	Propulsion/Avionics
PDF	Probability Distribution Function
PRD	Preliminary Requirements Document
RAMP	Risk Assessment and Management Program
RCS	Reaction Control System
RFP	Request for Proposal
S/W	Software
SE	Support Equipment
SEI	Space Exploration Initiative
Sh-C	Shuttle-C
SOFI	Spray-On Foam Insulation
SSF	Space Station Freedom
STAS	Space Transportation Architecture Study
STCAEM	Space Transportation Concepts and Analysis for Exploration Missions
STIS	Space Transportation Infrastructure Study
STS	Space Transportation System
STV	Space Transfer Vehicle
STVIS	Space Transfer Vehicle Information System
TCS	Thermal Control System
TEI	Trans-Earth Injection
TMI	Trans-Mars Injection
TPS	Thermal Protection System
TT&C	Telemetry, Tracking, and Control
TVC	Thrust Vector Control
TVS	Thermodynamic Vent System
UNIS	Unified Information System
USRS	Upper Stage Responsiveness Study
VCS	Vapor Cooled Shields
WTR	Western Test Range

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1.0 INTRODUCTION

With the initiative provided by the president to expand the exploration and habitation of space, a need arose to define a reliable and low cost system for transporting man and cargo from the earth surface or orbit to the surface of the moon or Mars. The definition of this system is two fold, the need for an low cost heavy lift Earth-To-Orbit system represents one of the major emphasis the other is the transportation system itself. The STV study has analyzed and defined an efficient and reliable system that meets the current requirements and constraints of both the existing and planned ETO systems as well as the surface habitation needs, as well arriving at the definition of key technologies needed to accomplish the these further needs. The results of the study provide a family of systems that support a wide range of existing and potential space missions. The simplest of the systems support the near earth orbital payload deliveries for both NASA and the DoD, requiring very short mission duration with no recovery of any portion of the system. The more complexity systems prove support for the interplanetary manned missions to both the moon and to Mars. These system represent state of the art systems that provide safety as well as reusable characteristics that allow the system to be used spaced based, the next step in the expansion of mans' presence in space.

The time to develop this STV family is now. Its role in complementing the space transportation infrastructure, keeps the United States of America as the world leaders in science, defense, and commercial space ventures for the 21 st century.

The space transportation tasks that the STV system must perform to transport humans with mission and science equipment from Earth to high earth orbits or the surfaces of the moon or Mars can be divided into three phases. (1) Transportation to-and-from low Earth orbit (LEO) being accomplished by the NSTS, ELVs, and new heavy-lift launch vehicles (HLLV) capable of 75 to 150 t cargo delivery; (2) space transfer vehicles providing round-trip transportation between LEO, lunar, and planetary orbits; and (3) excursion vehicles providing transportation between lunar/planetary orbits and their surfaces. Where one mode of transport gives way to another, transportation nodes can be utilized. In low Earth orbit, Space Station Freedom or a co-orbiting platform can serve that need. Elements of the space transfer and excursion vehicles are delivered by the HLLV and crews by the NSTS. Once all the elements have been delivered crews from SSF assemble, checkout, and then launch the vehicle. Following completion of the planned stay at the orbital node, lunar surface, or Mars, the transfer vehicles return the crew and a limited amount of cargo to LEO where the vehicles are refurbished and serviced for additional missions. Performing the transportation functions in this manner maximizes the commonality and synergism between the lunar and Mars space transportation systems and brings the challenge of the exploration initiatives within the reach of orderly technology advancement and development.

Our final report addresses the future space transportation need and requirements based on our current assets and their evolution through technology/advanced development using a path and schedule that supports our world leadership role in a responsible and realistic financial forecast. Always, and foremost, our recommendations place high values on the safety and success of missions both manned and unmanned through a total quality management philosophy at Martin Marietta.

2.0 SYSTEMS ENGINEERING AND REQUIREMENTS

The objective of the systems engineering task was to develop and implement an approach that would generate the required study products as defined by program directives. This product list included a set of system and subsystem requirements, a complete set of optimized trade studies and analyses resulting in a recommended system configuration, and the definition of an integrated system/technology and advanced development growth path. A primary ingredient in Martin Marietta's approach was the TQM philosophy stressing job quality from the inception. Included throughout the Systems Engineering, Programmatics, Concepts, Flight Design, and Technology sections are data supporting the original objectives as well as supplemental information resulting from program activities.

The systems engineering approach used a reference baseline from past study documentation to establish the foundation for further study, see Figure 2.0-1. The Design Reference Missions (DRMs) were derived from this reference database. These DRMs provided the primary bounding requirements for the development and definition of the three major study tasks; the system and subsystem requirements, the conceptual design, and the studies and analyses that supported the formulation of both the requirements and the design. Combined with inputs from the technology/advanced development effort, the products of these tasks included a recommended

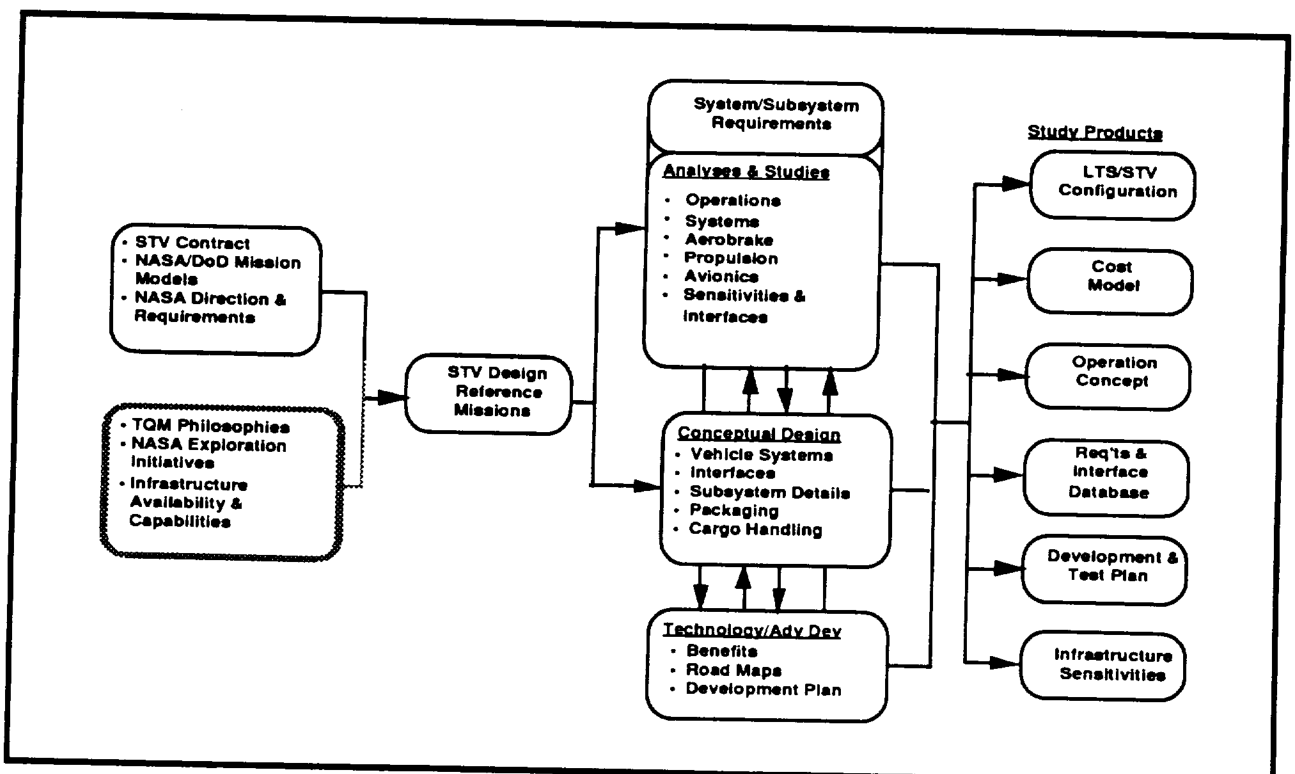


Figure 2.0-1 Systems Engineering Approach

LTS/STV configuration, a cost model, an operations concept for conducting manned lunar missions, system and subsystem requirements and interfaces database, a development and test plan, and defined infrastructure sensitivities. The basis for this approach was to ensure that each task was completely integrated with its appropriate interface activity, resulting in a product that will meet all program requirements, use the best of the technology community, and have the flexibility to change as the space infrastructure matures.

The primary result of the analyses and studies was the recommendation of a single propulsion stage LTS configuration that supports several different operations scenarios with minor element changes. This concept has the potential to support two additional scenarios with complex element changes. The space based LTS concept consists of three primary configurations - Piloted, Reusable Cargo, and Expendable Cargo (Fig. 2.0-2).

The piloted configuration has a central propulsion/avionics core, a crew module, six droptank assemblies, and an aerobrake that includes RCS, propellant tanks, and avionics modules. Vehicle

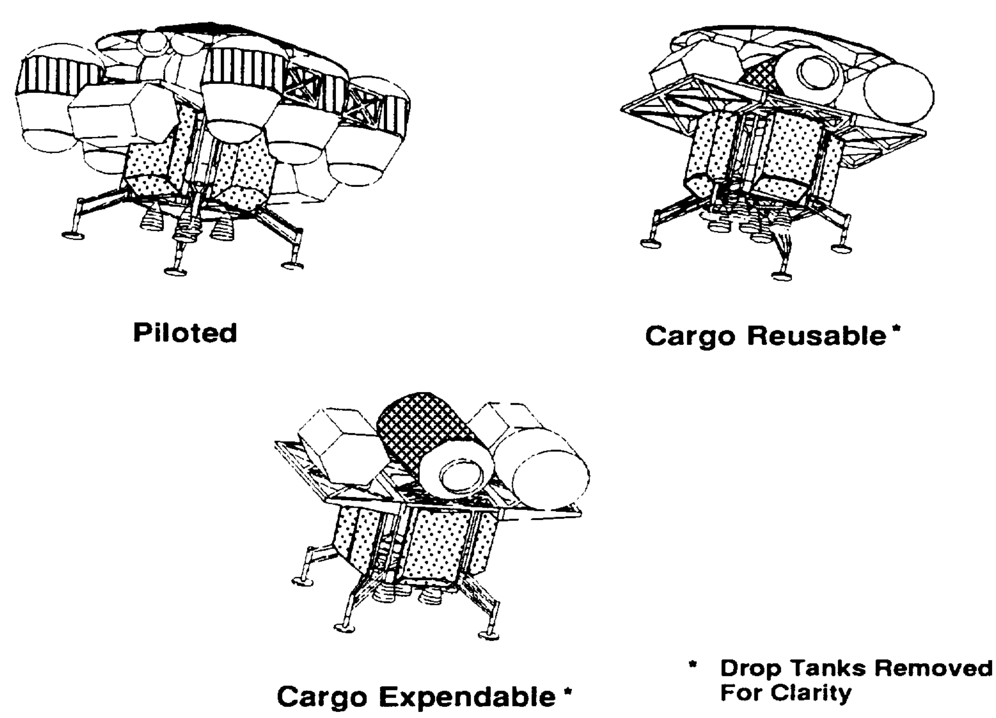


Figure 2.0-2 Piloted, Reusable Cargo, and Expendable Cargo LTS Configurations

dry mass is 27.6 tonnes and requires 174 tonnes of cryogenic propellant to perform a lunar mission delivering 14.6 tonnes of cargo and a crew of four to the lunar surface and returning the crew to the Low Earth Orbit (LEO) transportation node.

The reusable cargo configuration has a central propulsion/avionics core, six droptank assemblies, and an aerobrake that includes RCS, propellant tanks, and avionics modules. In accommodating the cargo, the crew module is removed and the large cargo platform is mounted in its place. Vehicle dry mass is 22.3 tonnes that requires 169.3 tonnes of cryogenic propellant to perform a lunar mission delivering 25.9 tonnes of cargo to the lunar surface and returning to the LEO transportation node.

The expendable cargo configuration has a central propulsion/avionics core, six droptank assemblies, and a large cargo platform mounted in place of the crew module and aerobrake. Vehicle dry mass is 18.8 tonnes that requires 146.5 tonnes of cryogenic propellant to perform a lunar mission delivering 33.0 tonnes of cargo to the lunar surface. The vehicle is then expended.

The configurations (Fig. 2.0-3), were derived from the single propulsion stage concept. These configurations represent an All-Propulsive Space Based configuration, an All-Propulsive Nonspace Based configuration, and a High Energy Upper Stage for use with an Heavy Lift Launch Vehicle (HLLV) or the LTS.

The All-Propulsive Space Based configuration is similar in make-up to the recommended LTS with the exception of the replacement of the aerobrake with additional propellant tanks to support the earth capture maneuver.

The All-Propulsive Nonspace Based configuration expends all the mission elements before return of the Apollo type ballistic cab to the Earth's surface. Additional analysis and study are required to develop the physical and functional details of these concepts.

Details on the High Energy Upper Stage will be provided in the flight design section of this document.

Additional analyses and studies of the systems that make up the LTS configuration (aerobrake, propulsion, avionics and structure) show key links to similar system functions in other planned infrastructure components such as the proximity operations vehicle and deep space exploration systems.

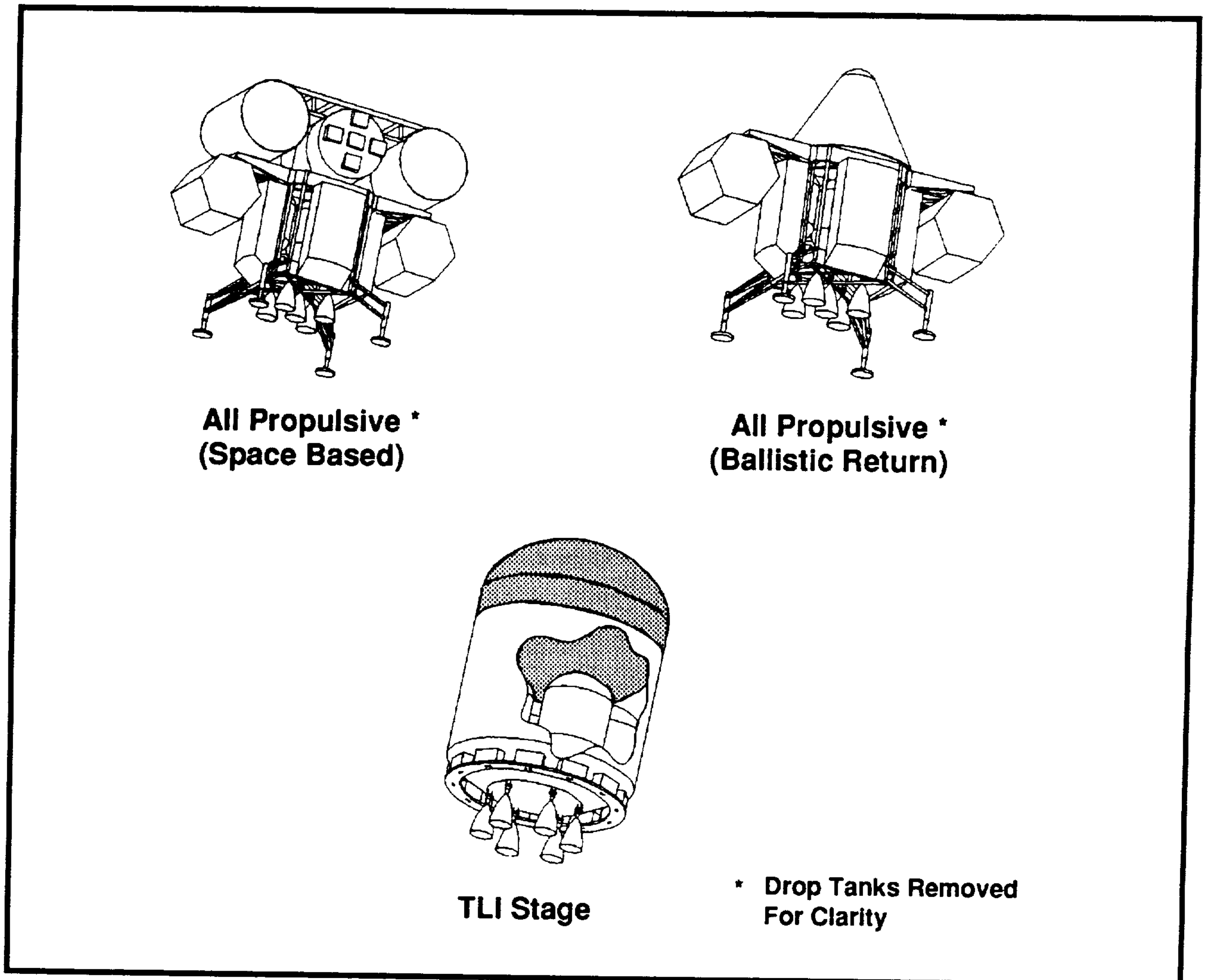


Figure 2.0-3 Alternative Configurations

2.1 DESIGN REFERENCE MISSIONS (DRMS)

The seven DRMs that have provided the bounding requirements and top level system characteristics for the development of the STV/LTS have been defined through the process illustrated in Figure 2.1-1. A comprehensive list of mission candidates was developed from several reference sources; the 1989/90 CNDB, supplemented with the STV augmented CNDB (09 Aug 1989), the 1989 Air Force Space Command National Mission Model, and the Human Exploration Study Requirements Document. This list was placed through a top level screen of several selection criteria that included mission quantity, payload compatibility, and IOC date. This resulted in a candidate list that included six near Earth missions, four lunar missions, three Mars missions, and four planetary exploration missions. The second and final screen resulted in nine missions being defined as STV DRMs. Five were near Earth missions; reduced to four with the release of the CNDB 90; two lunar missions; one Mars mission, and one planetary exploration mission, also deleted with the release of the CNDB 90. From these seven DRMs have come the bounding requirements and characteristics for the LTS/STV configurations (Fig. 2.1.2).

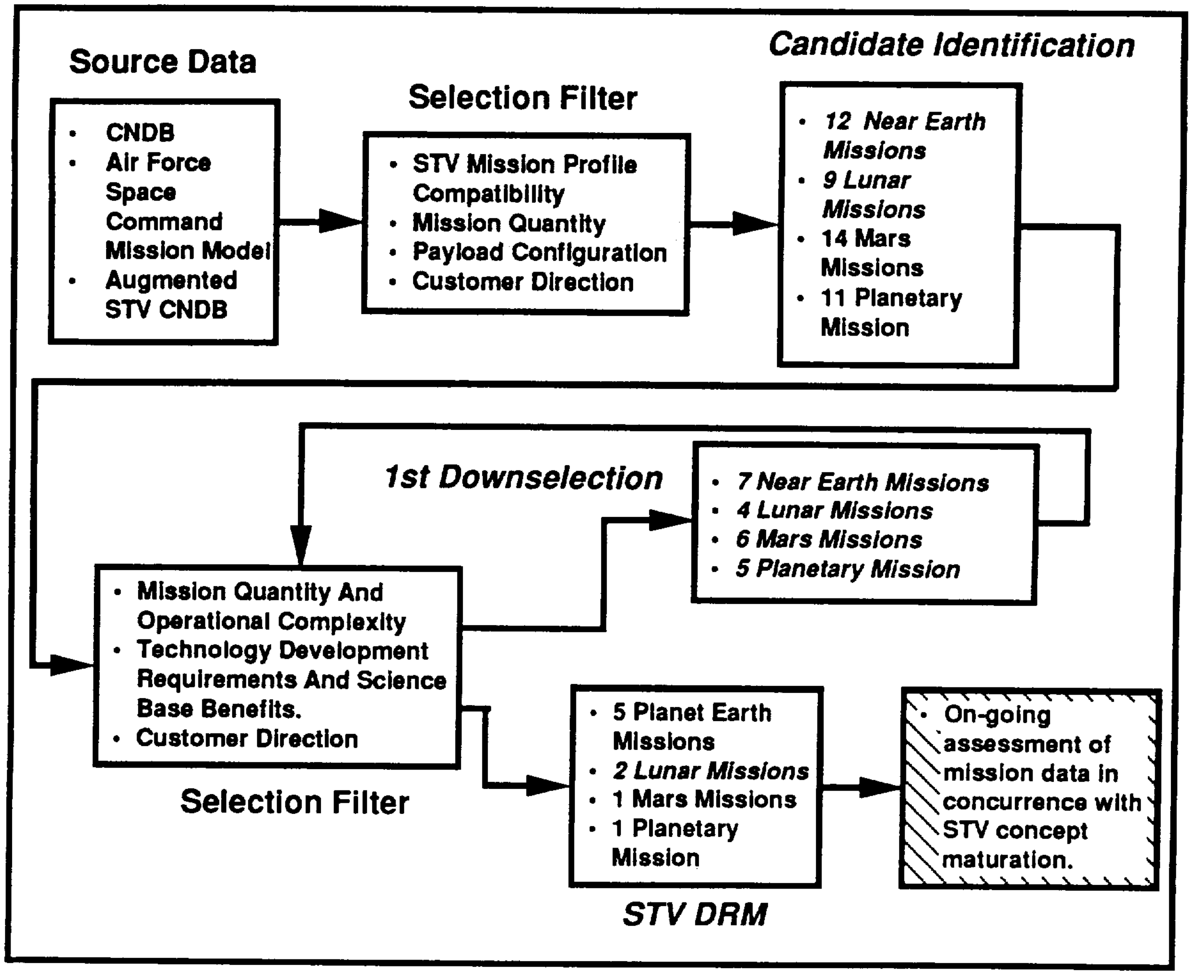


Figure 2.1-1 STV DRM Selection Process

Each mission provided a unique set of bounding requirements by which the STV system must be designed to perform. From this collection of requirements, seven were found to represent the largest impact to the development of the STV. These requirements included man-rated and reusable, payload type, payload mass, first flight, number of missions, duration of each mission, and the total mission Δ-velocity. Table 2.1-1 shows the interrelationship of these requirements across the overall range of STV DRMs.

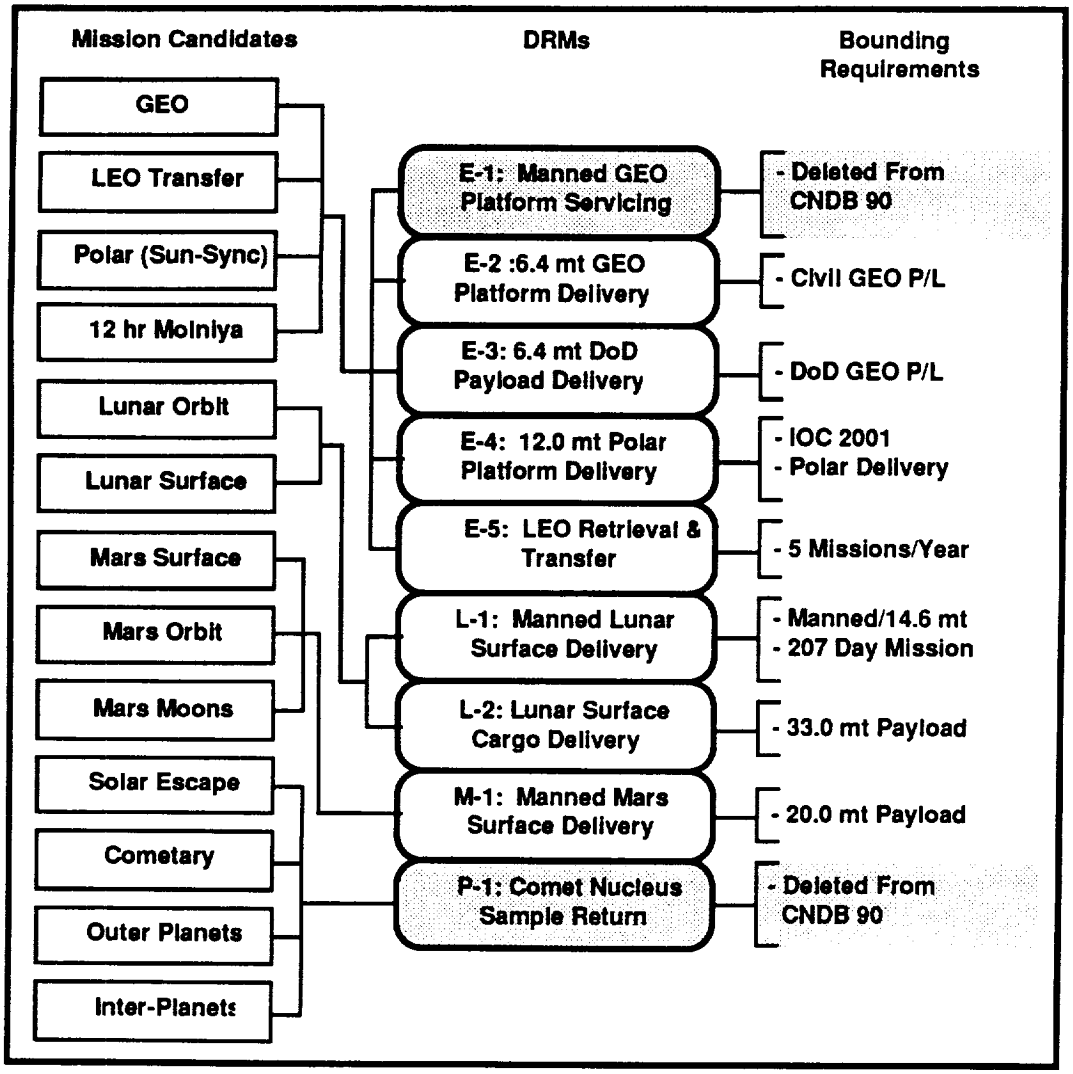


Figure 2.1-2 STV Design Reference Missions and Their Corresponding Requirements

2.1.1 Mission Characteristics

The seven selected STV DRMs represent three destinations; near Earth, lunar, and Mars. The following discussion provides detailed characteristics for missions going to each destination.

Table 2.1-1 Design Reference Mission Requirements Summary

DRM	Characteristics						
	Re-Usable	Payload	Mass	First Flight	Flight Quantity	Mission Duration (max.)	Delta-V (m/s)
L-1	Yes	Crew/Supplies	14.6 mt	2004	27	207 days	9500
L-2	No	Cargo	33.0 mt	2002	10	16 days	6425
E-2	No	Cargo	6.4 mt	2005	1	6.0 hrs	2900
E-3	No	Cargo	6.4 mt	2001	14	6.0 hrs	2900
E- 4	Yes/No	Equipment	12.0 mt	2001	4	TBD	6600
E-5	Yes	Lunar P/L	71 mt	1998 (ILC	~150	2 days	240
M-1	No	Mars Vehicles	20.0 mt	2015	12	50	4700

2.1.1.1 Lunar—The lunar surface mission delivers a four man crew and 14.6 tonnes of cargo and returns 0.5 tonnes of cargo to Earth orbit, or 33.0 tonnes of cargo to the lunar surface in an expendable mode. Major elements of the mission include trans-lunar injection, lunar capture, lunar surface descent and ascent, trans-Earth injection, and Earth orbit capture. Table 2.1.1.1-1 and Table 2.1.1.1-2 define the primary payload and operational characteristics associated with these missions. Crew environmental control, power, data communications, and life support requirements are provided by the transportation system during the transportation phases of the mission and by the surface support systems throughout the duration of the stay.

2.1.1.2 Near Earth—The Geosynchronous Earth Orbit (GEO) servicing mission initially defined as a STV DRM candidate was deleted from the final DRM set when it was eliminated from the CNDB 90. Two unmanned delivery missions were defined to deliver a civil and a DoD payload to GEO. These payloads consist of a satellite and a science platform both weighting 6.4 tonnes.

Tables 2.1.1.2-1 and 2.1.1.2-2 define the physical and operational characteristics associated with these missions. Major mission elements include GEO insertion, platform placement, and vehicle disposal. Data communications, as required by the payload, are provided by the transportation system.

Table 2.1.1.1-1 Lunar Surface Cargo Mission Characteristics

IOC/Technology Availability		2002
Payload	Classification	Cargo
	Mass	33.0 t (To Surface)
	Volume	TBD
	Thermal	Payload Provided
	Power	Payload Provided
	Fluid	None
	Data Communications	200 kb/s
Flight Rate	Responsiveness	N/A
	Nominal Rate (Flight/Year)	1
	Flight Quantity	10
Duration	Max Mission Time (Days)	16 days (TBR)
	Payload Support Time (Days)	16 days (TBR)
	Destination Stay Time (Days)	N/A
Destination	Accuracy	3 m R
	Destination	Lunar Surface
Delta-V		6403 m/s
Source		HE Study Req'ts (01 Dec 89)

Table 2.1.1.1-2 Manned Surface Mission Characteristics

IOC/Technology Availability		2002
Payload	Classification	Crew & Supplies
	Mass	14.6 t (To Surface)
	Volume	310 cu m (Cargo)
	Thermal	1kWt (Crew)/Payload Provided (Cargo)
	Power	2 kWe (Crew)/Payload Provided (Cargo)
	Fluid	None
	Data Communications	200 kb/s
Flight Rate	Responsiveness	30 Days (TBR)
	Nominal Rate (Flight/Year)	2
	Flight Quantity	27
Duration	Max Mission Time (Days)	207 days
	Payload Support Time (Days)	27 days
	Destination Stay Time (Days)	180 days (Max)
Destination	Accuracy	3 m R
	Destination	Lunar Surface
Delta-V		9721 m/s
Source		HE Study Req'ts (01 Dec 89)

Table 2.1.1.2-1 6.4 tonne GEO Platform Delivery Mission Characteristics

	IOC/Technology Availability	2005
Payload	Classification	GEO Platform
	Mass	6.4 t (22 Klbs)
	Volume	100 cu m (TBR)
	Thermal	Payload Provided
	Power	Payload Provided
	Fluid	None
	Data Communications	200 kb/s
Flight Rate	Responsiveness	N/A
	Nominal Rate (Flight/Year)	1
	Flight Quantity	1
Duration	Max Mission Time (Days)	6.0 hrs
	Payload Support Time (Days)	6.0 hrs
	Destination Stay Time (Days)	N/A
Destination	Accuracy	TBD
	Destination	GEO Insertion
	Delta-V	4300 m/s (TBR)
	Source	STV Augmented CNDB (09 Aug 89)

Table 2.1.1.2-2 6.4 tonne GEO (DoD) Payload Delivery Mission Characteristics

	IOC/Technology Availability	2001
Payload	Classification	DoD Payload
	Mass	6.4 t (14 Klbs)
	Volume	200 cu m (TBR)
	Thermal	Payload Provided
	Power	Payload Provided
	Fluid	None
	Data Communications	200 kb/s
Flight Rate	Responsiveness	N/A
	Nominal Rate (Flight/Year)	1
	Flight Quantity	14
Duration	Max Mission Time (Days)	6.0 hrs
	Payload Support Time (Days)	6.0 hrs
	Destination Stay Time (Days)	N/A
Destination	Accuracy	TBD
	Destination	GEO Insertion
	Delta-V	4300 m/s
	Source	AF Space Command Mission Model

The polar orbit mission delivers six 12.0 tonne unmanned platforms to polar orbit. This mission will be conducted out of the west coast, with a contingency scenario for launching from the east coast. Table 2.1.1.2-3 defines the physical and operational characteristics associated with this mission. Major elements of the mission include polar orbit insertion and platform placement operations with plane changes for LEO injection if flown from the east coast. Data communications, as required by the payload, are provided by the transportation system during the transportation phases of the mission and by the platform support systems throughout the duration of the polar platform stay.

The LEO payload retrieval/transfer mission transports space transportation elements and lunar/Mars payloads between shuttle cargo parking locations and LEO transportation nodes. Major elements of the mission include orbit transfers (220 NM to 160/280 NM to 220 NM) and payload docking and handling. Table 2.1.1.2-4 defines the physical and operational characteristics associated with this mission.

2.1.1.3 Mars

The Mars Surface Manned mission supports the placement of Mars transfer/excursion vehicles, a four man crew, and a 20 tonne payload in a trans-Mars trajectory. Table 2.1.1.3-1 defines the physical and operational characteristics associated with this mission. Major elements of the mission include trans-Mars injection, payload separation and injection vehicle disposal. Data communications, as required by the payload, are provided by the transportation system during the injection phase of the mission.

2.1.1.4 Planetary

The Comet Nucleus Sample Return (KOPFF) mission initially defined as a STV DRM candidate was deleted from the final DRM set when it was eliminated from the CNDB 90.

2.1.1.5 STV Bounding Requirements

Using the characteristics compiled from this seven DRMs, the following detailed set of bounding requirements for the performance and operation of the STV system was developed. It should be noted that the characteristics associated with the LEO Payload Retrieval/Transfer mission were not considered in the development of these since the mission did not levy critical design or performance criteria on the STV system, but will be accommodated by the operational system.

Table 2.1.1.3-1 Mars Surface Manned Delivery Mission Characteristics

	IOC/Technology Availability	2015
Payload	Classification	Mars Transfer/Excursion Vehicles
	Mass	20 t. (cargo to surface)
	Volume	TBD
	Thermal	Payload Provided
	Power	Payload Provided
	Fluid	None
	Data Communications	TBD
Flight Rate	Responsiveness	30 days
	Nominal Rate (Flight/Year)	1
	Flight Quantity	12
Duration	Max Mission Time (Days)	50 days
	Payload Support Time (Days)	50 days
	Destination Stay Time (Days)	N/A
Destination	Accuracy	TBD
	Destination	Mars Orbit Insertion Trajectory
	Delta-V	4700 m/s (TBR)
	Source	HE Study Req'ts (01 Dec 89)
	Source	STV Augmented CNDB - 1989

Table 2.1.1.2-4 LEO Payload Retrieval/Transfer Mission Characteristics

	ILC/Technology Availability	1998
Payload	Classification	Lunar Payload
	Mass	71 t (TBR)
	Volume	400 cu m (TBR)
	Thermal	Payload Provided
	Power	Payload Provided
	Fluid	None
	Data Communications	TBD
Flight Rate	Responsiveness	N/A
	Nominal Rate (Flight/Year)	6
	Flight Quantity	60 (TBR)
Duration	Max Mission Time (Days)	2 days
	Payload Support Time (Days)	2 days
	Destination Stay Time (Days)	N/A
Destination	Accuracy	TBD
	Destination	SSF (220 x 220 nmi)
	Delta-V	240 m/s
	Source	HE Study Req'ts (01 Dec 89)

1) **First flight shall occur in 2001:** Across all missions, the IOC date is the primary driving requirement with first flight STV capabilities ranging from 2001 to 2006. The Mars manned mission was excluded from this assessment since, with its IOC date of 2016, there are no contributing impacts to the development of the system. Contained within this requirement are the impact and integration of technology, scheduling of the DT&E test program, and support node (i. e SSF) availability.
2) **Provide a total Δ-velocity up to 9.5 km/s:** With a Δ-velocity ranging from 9.5 to 2.9 km/s there is a direct correlation to vehicle sizing, ETO interfaces and performance, support node accommodation, and the propulsion system.
3) **System shall be capable of injecting a payload mass of up to 33 tonnes:** Combined with the performance requirements of 9.5 to 2.9 km/s the mass delivered defines vehicle sizing and structural configuration, support equipment, and directly influences the system operational cost.
4) **Mission durations of up to 50 days of full up operations and the capability of maintaining system operations for 207 days shall be accommodated.** Operational time impacts to the development of the STV system are constrained primarily to the manned missions, although the actual operating time for these missions is similar to the remaining STV missions. Of the 207 days required for the manned lunar mission, only 30 days of full up operations are needed, and for the manned Mars missions, 50 days are required.
5) **150 payloads, manned and unmanned, shall be delivered through the life of the system:** Quantities of payloads delivered will only impact the STV system infrastructure by influencing the economics of the development, operation, and recurring system cost.

Of the seven STV DRMs, the lunar missions (both manned and unmanned) represent the primary contributor to the STV growth requirements. To ensure the proper implementation of these requirements, the emphasis during the system concept definition and development phases focused on the lunar missions, with evolutionary considerations given to the GEO, planetary, and Mars missions.

2.2 REQUIREMENTS

Using the bounding requirements established by the STV DRMs, a set of system level requirements has been developed (Fig. 2.2-1). These requirements comprised basing, man-rating, maintenance and service life, Earth return, propellant, autonomy, and operations and interfaces. A portion of these requirements has been imposed either through NASA documentation, on-going studies, or the STV contract SOW. These derived requirements are the result of the system and configuration trades and analyses that have been conducted.

2.2.1 System Requirements

The requirements developed during the STV Concepts and Requirements Study were defined in two categories - general requirements that were imposed on systems supporting all transportation scenarios and mission unique requirements that impact specific missions such as lunar and Mars. The primary contributors to the development of these requirements have been the SEI Option 5 Human Exploration Requirements Document, the STV DRM bounding requirements (Section 2.1), STV studies and analysis results, and past transportation vehicle concept definition studies that include the OTV Concept Definition and System Analysis Study.

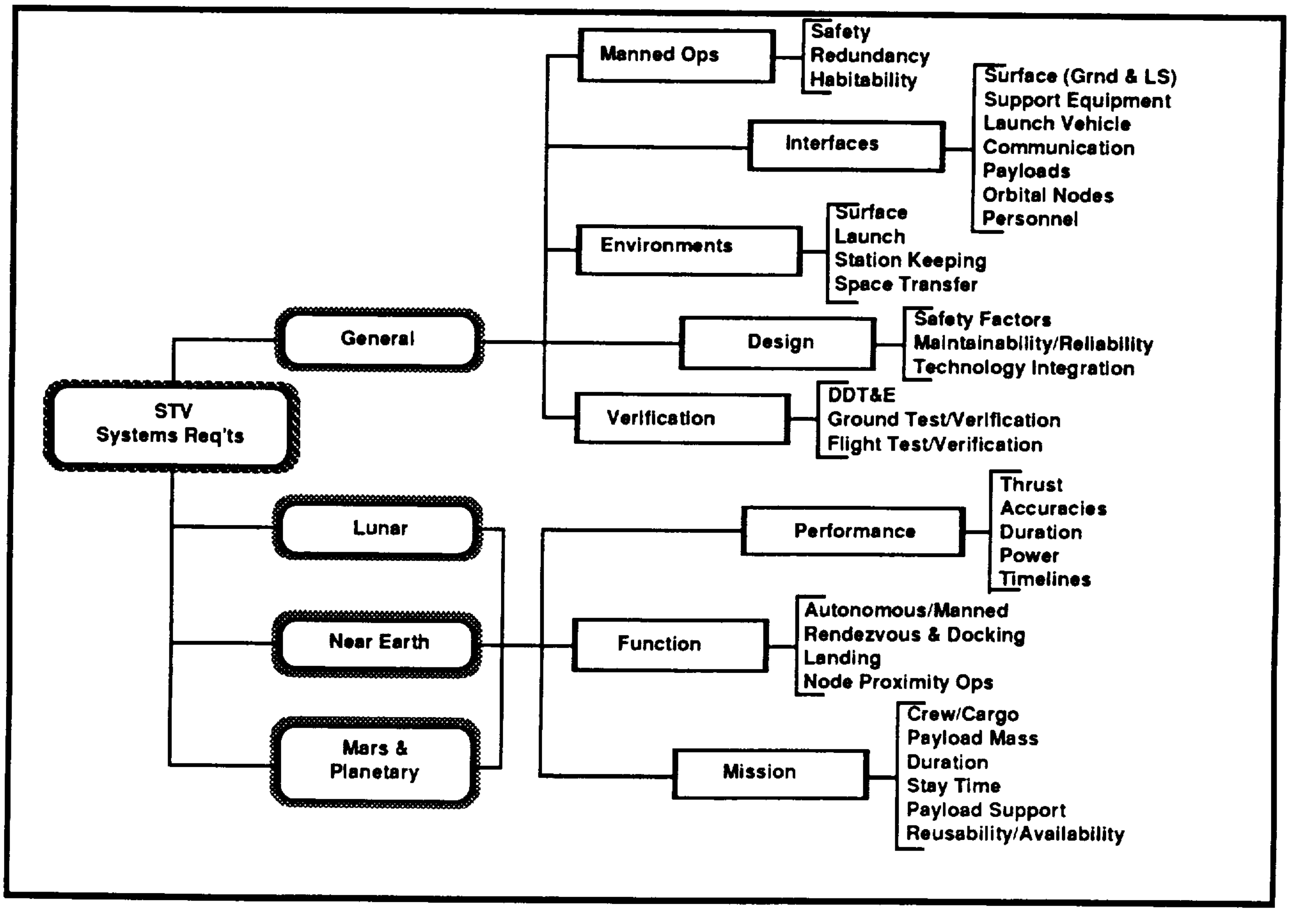


Figure 2.2-1 STV System Requirements

The overall program requirements, shown below, were derived from the requirements provided by NASA for the SEI analyses. These have been augmented to support the STV program definition of the goals of the transportation system. The STV/LTS shall:

a) Support the expansion of the human presence in GEO, on the Moon, and on Mars. The system shall deliver platforms and support their servicing. A derivative of the space transportation configuration shall retrieve and transfer payloads in LEO proximity. The objectives of the lunar missions are to deliver and return a crew and cargo to the lunar outpost and support further exploration of the lunar surface. The objectives of the Mars missions are to deliver and return a crew and cargo to the Mars outpost and to support further exploration of Mars.

b) Provide a transportation system capable of evolving between GEO missions, lunar missions, and Mars missions. These capabilities shall exist to derive transportation system alternatives to support varying exploration and budgetary strategies.

c) Obtain the knowledge and experience to enhance advancements in technology from the study and analysis of the transportation configuration and operations.

d) Maintain the safety and health of the crew throughout all missions. Protection shall be provided to eliminate forward and back contamination.

2.2.1.1 General Requirements—General STV requirements define manned operations, interfaces, mission environment, design, and verification. These will be imposed on all configurations and operations of the STV system.

2.2.1.1.1 Manned Mission Operational Requirements—The STV shall be capable of transporting personnel (one or more) to a safe haven. Mission abort trajectories shall provide for free return aborts for manned missions and planetary surface impacts for the disposal of unmanned mission hardware.

Crew Health & Hazard Systems shall conform to NASA STD-3000. Real-time monitoring, caution and warning, certification and revitalization of the atmosphere and water shall be provided. Pressurized crew facilities shall provide an atmosphere with a combination of pressure and oxygen achieving an equivalent 21% oxygen, 14.7 psi atmosphere within flammability and EVA constraints. Radiation requirements for the transfer vehicles and applicable to GEO and landing vehicles shall adhere to the "Guidance on Radiation Received in Space Activity". The capability to protect, maintain, and monitor the health and performance of the crew shall be provided.

A minimum of two crew members shall perform each scheduled EVA. Suit pressure/pre-breathe combinations for EVA shall achieve an R value of 1.22. In space and surface EVA provisions shall be made for each crew member. Providing simultaneous capability for the entire crew is not required.

Crew accommodations and human factors shall comply with NASA STD-3000. Training and simulations to maintain crew performance during long-duration, low and variable-g exposures shall be provided as appropriate. Crew habitats shall provide facilities for meals and recreation.

2.2.1.1.2 Interfaces—The transportation system shall interface with Earth based facilities, ground transportation systems, power systems, payload handling mechanisms, thermal management systems, and launch elements.

Ground operations will provide processing, assembly and checkout, and launch of space transportation elements. The STV elements shall be assembled and verified before ETO vehicle integration at a payload processing facility. Integration with the ETO vehicle shall be performed at the Vehicle Assembly Building (VAB). Assembly building/launch pad transfer shall be performed by a mobile transport system. ***Propellant will be loaded at the launch pad.*** The STV crew will be processed as part of an STS (or equivalent) mission launched with existing ground elements.

The Earth-to-Orbit system shall provide the hardware systems and/or support system that provides the capability for transportation elements to be delivered to LEO.

a) STV elements shall be delivered to a 160 nmi circular by 28-1/2 to 56° inclination orbit: STV IMLEO consists of 9300 tonnes (TBR: 4075 tonnes - lunar/GEO, 5200 tonnes - Mars) between 2001 and 2026. Between 2000 and 2014, a maximum IMLEO of 275 tonnes/year of vehicle components, payloads, and propellant, is required with a maximum of 900 tonnes/year from 2015 to 2026 required,
b) Payload diameters up to and including 10 m will be delivered to LEO,
c) A maximum of six ETO flights/year will be allocated to support space transportation missions,

d) Launch environments: Induced environments shall not exceed those of the STS or Titan IV,
e) Inline and/or kneel/trunnion attach point shall be accommodated,
f) A 1 kb/s data communication link shall provide for payload monitoring,
g) Thermal protection shall be comparable with Titan,
h) 5 kWt of power shall be provided to the payload,
i) Launch pad fluid management shall include propellant and vehicle purge and direct payload RF communications,
j) Crew: STS or equivalent shall provide manned transportation to the LEO node.

The Low Earth Orbit (LEO) transportation node shall provide the hardware systems and/or support systems that provide the capability for assembly, storage, checkout, refurbishment, and control transportation elements. An enclosure shall be provided to support the storage and mission preparation activities. Propellant management and storage shall be capable of providing a maximum storage time for a quantity not to exceed 174 mt, for 90 days. Propellant venting and disposal shall be accomplished without impact to the node. Launch of the STV shall be controlled and monitored by the node. EVA and automation capabilities shall be capable of supporting mission preparation activities. Crew ingress and egress shall support:

a) Transportation system element assembly and checkout,
b) Crew transfer from the transportation node to STV.

Transportation systems shall interface with all destination support elements. Manned systems shall interface with power systems, data systems, payload handling mechanisms, thermal and propellant management systems, life support systems, and launch elements. Unmanned systems shall interface with power systems, data systems, and payload handling mechanisms.

Systems shall be compatible with both Earth and space based voice and data communication systems. Cargo elements shall provide appropriate interfaces such as rigid attachment points and data communications.

2.2.1.1.3 ***Environments (Physical and Natural)*** —Natural environments for all STV mission destinations are defined by OEXP Study Data Book - FY89 Studies: Radiation, Moon, Mars, Phobos, Deimos, Trajectories, Human Factors and Performance, and Systems. Crew environments are defined by NASA-STD-3000: Anthropometry and Biomechanics, Human Performance Capabilities, Nature and Induced Environment, Crew Safety, Health Management, Architecture, Maintainability, and EVA.

2.2.1.1.4 Design Requirements—

2.2.1.1.4.1 Factors of Safety—

a) General factors of safety for metallic flight structures shall be verified by analysis and static test; Yield = 1.10, Ultimate = 1.50.

b) General factors of safety for nonmetallic flight structures shall be verified by analysis and static test; Nondiscontinuity areas = 1.50, Discontinuity areas and joints = 2.00.

c) General factors of safety for pressure:

- Windows, doors, hatches, etc., internal pressure only,
- Proof pressure = 2.00 x Mean Operating Pressure (MOP),
- Ultimate pressure = 3.00 x MOP,
- Engine structures and components,
- Proof pressure = 1.20 x MOP,
- Ultimate pressure = 1.50 x MOP,
- Hydraulic and pneumatic systems,
  a. Lines and fittings, less than 1.5 inches (38 mm) diameter
    - Proof pressure = 2.00 x MOP,
    - Ultimate pressure = 1.50 x MOP,
  - b. Lines and fittings, 1.5 inches (38 mm) dia. or greater
    - Proof pressure =1.20 x MOP,
    - Ultimate pressure = 1.50 x MOP,
  - c. Reservoirs/pressure vessels
    - Proof pressure = 1.10 x MOP,
    - Ultimate pressure = 1.50 x MOP,
  - d. Actuating cylinders, valves, filters, switches
    - Proof pressure = 1.50 x MOP,
    - Ultimate pressure = 2.00 x MOP,
- Personnel compartments, internal pressure only
  - Proof pressure = 1.50 x MOP,
  - Yield pressure = 1.65 x MOP,
  - Ultimate pressure = 2.00 x MOP.

Failure Tolerance—Fault detection/fault isolation and reconfigurations of critical systems will be provided (ref. 3: NHB 53000.4 (1d-2) "Safety, reliability, Maintainability and Quality Provisions For The Space Shuttle Program"). All mission critical failures shall be detected.

Redundancy for man-rated elements shall be dual-fault tolerant (Fail-Op, Fail-Op, Fail-Safe). Electrical systems redundancy shall be; Fail-Op, Fail-Op, Fail-Safe. (ref. 3: NHB 53000.4 (1d-2) "Safety, reliability, Maintainability and Quality Provisions For The Space Shuttle Program"). Critical mission support functions shall be one failure tolerant. Critical functions affecting crew safety and survival shall be two failure tolerant. Meteoroid impact failures shall not endanger the crew or mission survivability of the mission. Pyrotechnic system redundancy shall be; Fail-Safe, Fail-Safe. (ref. 3: NHB 53000.4 (1d-2) "Safety, reliability, Maintainability and Quality Provisions For The Space Shuttle Program"). Mechanical system redundancy shall be; Fail-Op, Fail-Op, Fail-Safe. (ref. 3: NHB 53000..4 (1d-2) "Safety, reliability, Maintainability and Quality Provisions For The Space Shuttle Program").

The service life of STV systems and subsystems shall be a minimum of five missions. Drop tanks, replacement of consumables, and aerobrake shall be excluded. There will be no scheduled in-flight maintenance. All scheduled maintenance shall take place at the Earth transportation and space based nodes. Removal and replacement shall be done at the functional component level. Non-pressurized systems shall be accessible to telerobotic or EVA maintenance. In-flight systems shall diagnose failures, distinguish between sensed and "real" failures, perform adequately with redundant sensors, identify failures to the replacement level, automatically compensate for failures, notify the crew of proper operational and maintenance procedures, and provide thorough on-line maintenance documentation. The capability shall exist to perform unscheduled maintenance on flight/life support elements during all phases of the mission.

2.2.1.1.5 Technology—First flight shall not be impacted by technology development schedules. System architecture will allow incorporation of new technologies as they become available.

2.2.1.1.6 Verification—Overall reliability shall be demonstrated and verified by testing (ref. NHB 53000..4 (1d-2) "Safety, reliability, Maintainability and Quality Provisions For The Space Shuttle Program"). Requirement verification shall be performed either by analysis or test. System shall be certified for flight only after the requirement verification has been satisfactorily completed. All critical mission elements shall be verified by flight test. All critical mission elements shall be verified by ground test to the extent practical.

2.2.1.2 Lunar Mission Requirements—All lunar deliver and return mission shall use LLO. There shall be no direct landing flights. Piloted mission shall obtain Low Lunar Orbit (LLO) (300 km circular) before descent and Trans-Earth Injection (TEI). Transportation elements deployed in

LLO shall be stable. The transportation system shall deliver 429 tonnes of PSS elements to the lunar surface between 2002 and 2026, 142.8 tonnes between 2002 to 2007, 106.0 tonnes between 2008 to 2013, and 189.9 tonnes between 2014 to 2030.

Piloted flights shall deliver a crew of four and a maximum of 14.6 tonnes of cargo to the lunar surface and return a crew of four and a maximum of 0.5 tonnes of cargo to Earth orbit. Cargo flights shall deliver a maximum of 33.0 tonnes of PSS components. The flight rate for the delivery of these payloads shall not exceed one mission per year.

The transportation system shall be capable of autonomous rendezvous and payload propellant transfer. This shall include the capability for unmanned operations. The transportation vehicle shall aerobrake at Earth return with an entry velocity limited to 11.1 km/s. The aerobrake shall be removable for expendable missions. The transportation system shall be capable of landing on the lunar surface on a 50 meter diameter pad, level within 2 deg (improved) or on unimproved landing pad level within 15 deg. Landing shall also be accomplished over surface irregularities not more than 1 meter in height (unimproved). Landing on the lunar surface shall be within a three meter radius of a surface beacon (unimproved).

The transportation system shall be capable of supporting mission operations that shall not exceed a planned duration of 4360 hours (180 days) from Earth launch to Earth return. All system elements shall remain in lunar proximity during manned occupation. The lunar surface transportation system shall be compatible with a lunar surface stay time of 4360 hours (180 days) of which 96 hours shall be without surface support. The transportation system shall provide support to the transported cargo for a maximum of 2400 hours. This period includes 48 hours following landing and before ascent.

During flight, the system shall provide $1kW_t$ for crew thermal control by maintaining a sufficient window from the cargo modules to space, $2kW_e$ for crew with no cargo support, and 200 kb/s data rate for health and status monitoring.

Using the following requirements, the transportation system shall provide performance capabilities of delivering crew and cargo.

a) Propulsion system utilizes cryogenic propellant,
b) Two engines out will not abort the mission,
c) Total cryogenic boil-off shall not exceed 2% per month,
d) 1% reserves for Isp,

e) 1.5% residual,

f) 5% ullage,

g) Transportation shall be sized to accomplish:

Flight Phase	Δ-Velocity (m/s)	Phase Duration (days)
Pre-Injection Preparation	10	3
Trans-Lunar Injection	3259	0.1
Trans-Lunar Coast	10	3
Lunar Orbit Insertion	1098	0.1
Lunar Orbit Operations	25	0.5
Pre-deorbit Preparation	5	1
Deorbit to Landing	2000	0.1
Surface Operations	-	180
Ascent to Orbit	1900	0.1
Lunar Orbit Operations	25	0.5
Trans-Earth Injection	1098	0.1
Trans-Earth Coast	10	3
Earth Orbit Insertion	275	0.1
Earth Orbit Operations	40	2

An unmanned mission does not require meteoroid/debris protection. In-space propellant transfer shall be performed between the vehicle and LEO node, internal vehicle tankage, and the vehicle and PSS support equipment on the lunar surface. The transportation system shall require no major refurbishment in space. Assembly activities, vehicle servicing, and maintenance will occur at the LEO node and the lunar surface.

The first manned flight shall support manned occupation on the lunar surface by 2004. All proximity operations are directly viewable by the crew. One man operation shall be provided for contingencies. Crew module accommodations shall include;

a) Accommodations for a crew of four with provisions for eight in an emergency,

b) Powered operations for 9 days,

c) Protection against meteoroid and radiation,

d) Waste water management,

e) Two egress routes,

f) Automatic vehicle controls with manual override .

The first cargo flight will be to the lunar surface by 2002. Multiple payloads shall be accommodated. Cargo flights shall be configured for expendable and reusable operation.

2.2.1.3 Near Earth Mission—The transportation system shall be capable of delivering payloads to LEO between 2001 and 2030, GEO between 2001 and 2019, and to a polar orbit between 2001 and 2008. Missions shall deliver a maximum of 12.0 tonnes with a flight rate not exceeding two missions per year. The system will be capable of autonomous rendezvous, docking, and payload/propellant transfer. Reusable configurations will use an aerobrake return to LEO. The transportation system shall be capable of supporting mission operations that do not exceed 2 days in duration from Earth launch to payload delivery and return.

During flight the system shall provide no cargo support except for 200 kb/s data rate for health and status monitoring.

Using the following requirements, the transportation system shall provide performance capabilities of delivering cargo to GEO, LEO, and a polar orbit.

a) Propulsion system utilizes cryogenic propellant,
b) Single engine out will not abort the mission,
c) Total cryogenic boil-off shall not exceed 2% (TBD) per month,
d) 1% reserves for Isp,
e) 1.5% residual,
f) 5% ullage.

Meteoroid/debris protection shall not be provided for unmanned near Earth configurations. In-space propellant transfer is performed between the vehicle and LEO node and internal vehicle tankage. The transportation system will not require major refurbishment in space. Assembly, vehicle servicing, and maintenance activities will occur at LEO node when required.

The first cargo flight is to a near Earth destination by 2001. Provisions for multiple payload shall be accommodated. Cargo flights shall be configured for expendable and reusable operation.

2.2.1.4 Mars Mission—The system shall be capable of supporting the delivery of 20 tonnes of cargo and a crew of four to the Mars surface between 2015 and 2026.

2.2.2 Key Design Drivers

As the description of the LTS/STV configuration matured, eight system requirements were found to be key design drivers. The impacts of these requirements to the design of the system are defined below. It should be noted that a change in any one of these requirements has the potential of completely altering the results of the configuration selection activity.

System shall deliver 14.6 tonnes of cargo and a crew of four to the surface and return. Delivery of 14.6 tonnes of cargo and a crew of four represents the maximum propellant requirement of the three mission scenarios (piloted, reusable cargo, and expendable cargo). Transforming the piloted system to an expendable cargo configuration provides the capability of delivering 37.4 tonnes of cargo with the same propellant tanks as carried on the piloted mission. Sizing the propellant tanks and vehicle for the 33.0 tonne cargo mission will result in a cargo capability well short of the 14.6 tonne requirement in the piloted mode.

System shall be reusable for a minimum of five missions: System reuse requires return of more of the vehicle elements to a LEO node to make the scenario economically feasible. To support this, the IMLEO required for the mission increases to support the return performance requirements. A LEO node becomes the primary support element for assembly, checkout, and verification. To minimize the assembly requirements at the LEO node, quick disconnects are required in major system elements, impacting IMLEO as well as driving technology requirements. Within the vehicle itself system health monitoring and aeroassist become mandatory minimizing performance requirements and LEO node maintenance. While reducing LEO node EVA/IVA requirements, the additional avionics equipment increases the IMLEO.

Manned systems shall be fault tolerant. Increasing the avionics complexity to comply with this dual-fault tolerant requirement adds additional mass and is second only to the propellant as the major contributor to the IMLEO. This added complexity requires additional software. These additions become enabling technology and will have a direct impact on system availability.

System shall deliver 429 tonnes to the lunar surface between 2004 and 2030 as defined by the PSS requirements document (05 Jun 90): Compliance with the manifest delivery schedule defined by PSS requires the use of a minimum of four expendable cargo missions as shown in Figure 2.2.2-1. Minor reallocation of the cargo can significantly reduce the LCC costs of the LTS/STV program by allowing the reuse of three of these four cargo vehicles.

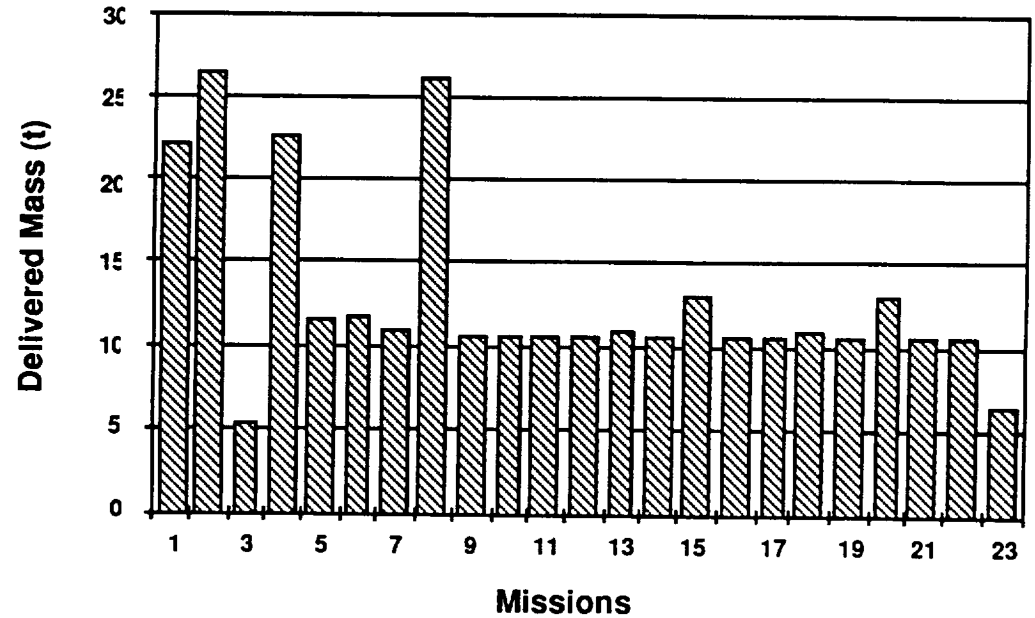


Figure 2.2.2-1 PSS Manifest Lunar Surface Delivery Requirements

The large cargo requirements in these expendable missions translate into major impacts to support systems such as KSC, the LEO node, and the handling of the cargo once delivered to the surface.

Space Station Freedom shall be used as the LEO transportation node. With SSF used as the LEO node, all interfaces with the supporting space infrastructure (KSC, ETO, PSS, and others) and the LTS/STV must be shared with those on SSF. This increases the LTS IMLEO since the SSF interfaces have been designed for stationary operations where weight restraints do not pay as much of a penalty as they do on a transportation vehicle. The handling and storage of propellant tanks have physical and safety impacts. Present data shows that the crew requirements for assembly and servicing of the LTS/STV fleet range from 400 to 1200 manhours or at a maximum 70% of the available crew time at SSF (Fig. 2.2.2-2). Contamination issues must be addressed to ensure that the SSF environment is not adversely affected. If the management and control of contamination falls on the LTS side of the interface, the potential exists for significantly increasing the IMLEO of the system.

System IOC shall be 2001 with initial manned flight in 2006: To support a mission in 2001, necessary technology must be at Level 6 or at PDR maturity by 1996. Based on current technology plans, the potential for the highly advanced systems necessary to meet the requirements of the STV/LTS program is moderate at best.

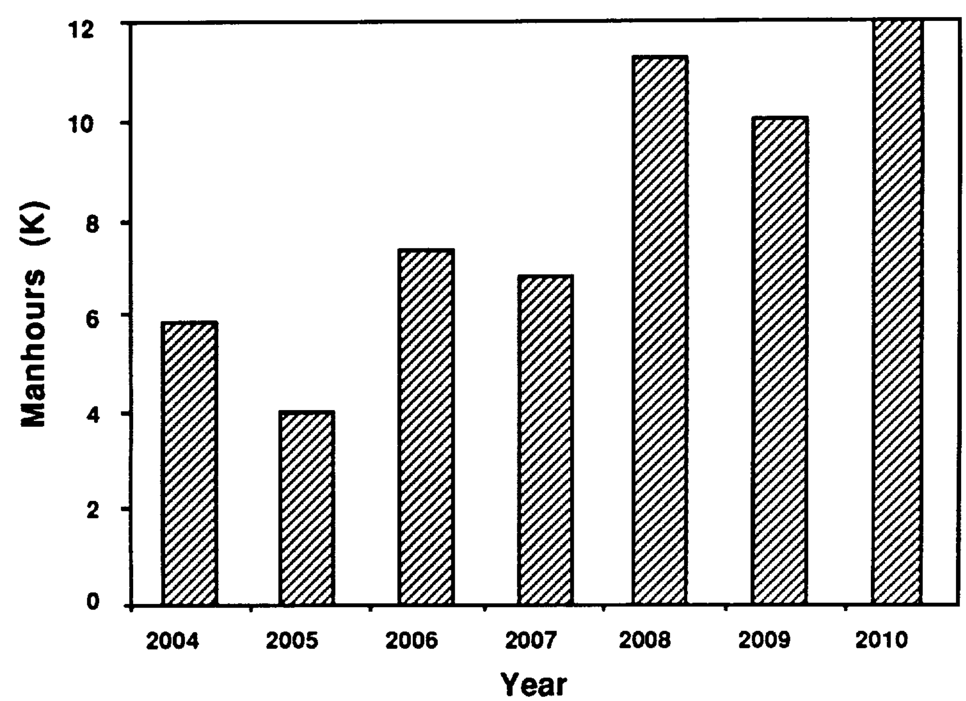


Figure 2.2.2-2 STV Assembly and Support Manpower Requirement

Propulsion system shall utilize LO_2/LH_2 propellant: Cryogenic propellants require complex and expensive storage equipment both at LEO and the lunar surface. Development and transportation of this equipment directly impacts the STV/LTS economically and physically. Replacement of the cryogenic propulsion system with an advanced propulsion system, such as a nuclear thermal rocket (NTR), can increase the mass capability to the lunar surface by as much as 100%, as shown in Figure 2.2.2-3. This translates into a lower IMLEO if the current PSS mass requirements are maintained.

System shall be capable of autonomous operation: Increasing the avionics complexity to provide autonomy adds additional mass second only to the propellant as the major contributor to the IMLEO. Included in this complexity is the required additional software. With this requirement, software becomes an enabling technology having a direct impact on system availability. Training requirements and facilities for the flight crews are reduced by implementing autonomous operations.

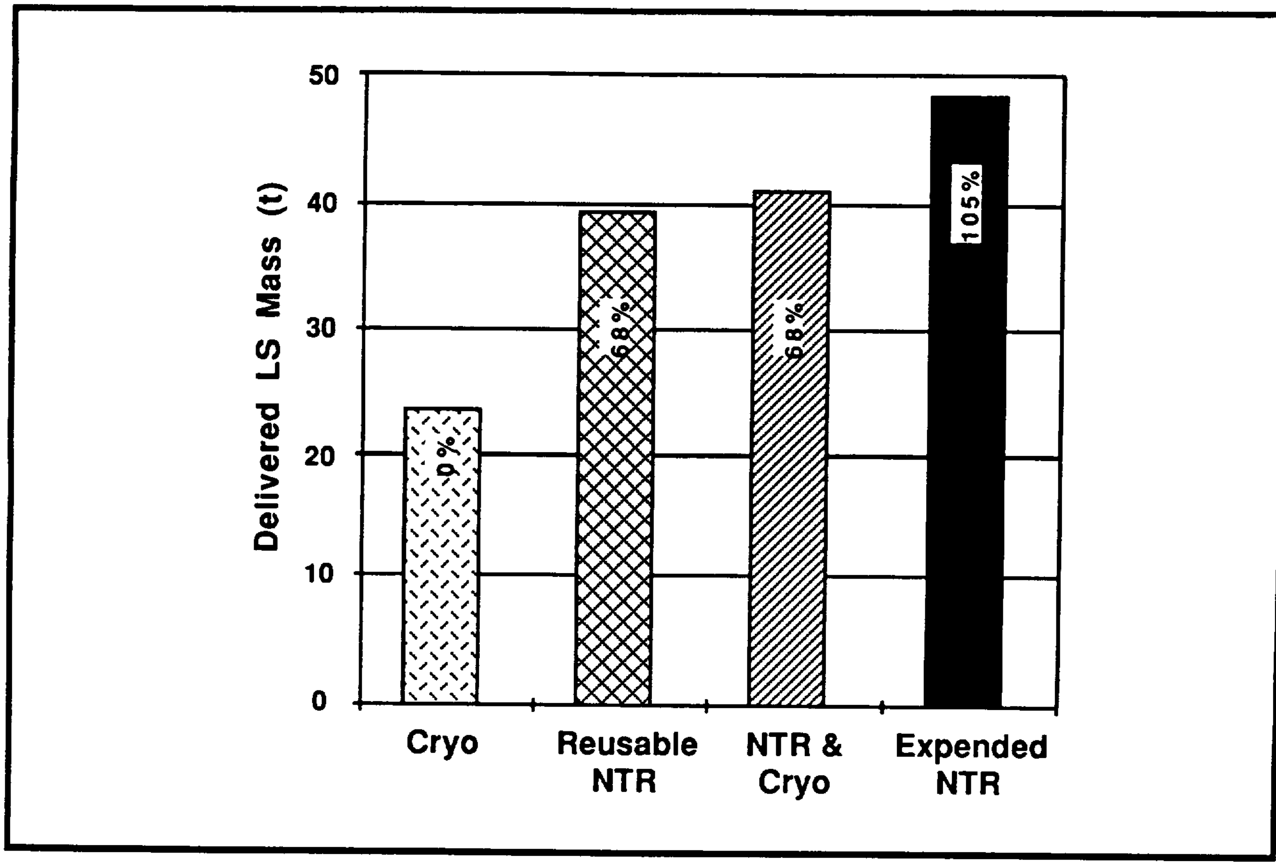


Figure 2.2.2-3 Propulsion System Capabilities to the Lunar Surface

2.3 SYSTEM TRADE STUDIES & ANALYSES

This section describes the objectives, analyses and results of top level systems trades performed to define and select the optimum STV concept or family of vehicles. Top level program decisions were made regarding aeroassist versus all propulsive, vehicle growth options, performance impact of lunar liquid oxygen, direct descent versus lunar orbit, etc. The results of substantiating system trades are included in this section following the description of the STV concept selection process.

2.3.1 Approach

The analysis and study activities of the STV study program were made up of six major areas; systems, operations, avionics, aerobrake, propulsion, and interfaces, as defined in Figure 2.3.1.-1 These categories were defined within the original proposal and updated in the initial phases of the program with inputs from our MFSC customer as well as ongoing studies. The primary emphasis of the activity has been in the systems and operations areas from which the data on the recommended configuration and its operation and performance evolved. Results of the system and flight operations efforts were used to conduct detailed analyses and studies on the subsystems as defined and constrained by the recommended system. Although shown as a one-way street as far as the flow of data, this was an integrated process with all subsystem data passed back through systems and operations to ensure compliance.

2.3.2 Mission Operations Analysis

The STV study program addressed the elements of the STV systems from receipt of hardware items at KSC to the disposal of operational items at the completion of its mission. The mission operations categories included ground, orbital, flight, and surface studies with the emphasis placed on supporting the Option 5 lunar outpost missions. The mission study traded the use of the lunar, near Earth, or planetary missions as the primary driver in the development of the STV configuration. Results were largely influenced by Martin Marietta's involvement in the MSFC "Skunk Works" effort. Since the primary focus of the "Skunk Works" was the lunar missions, the bulk of the data available supported the continuation of the detailed definition and description of a

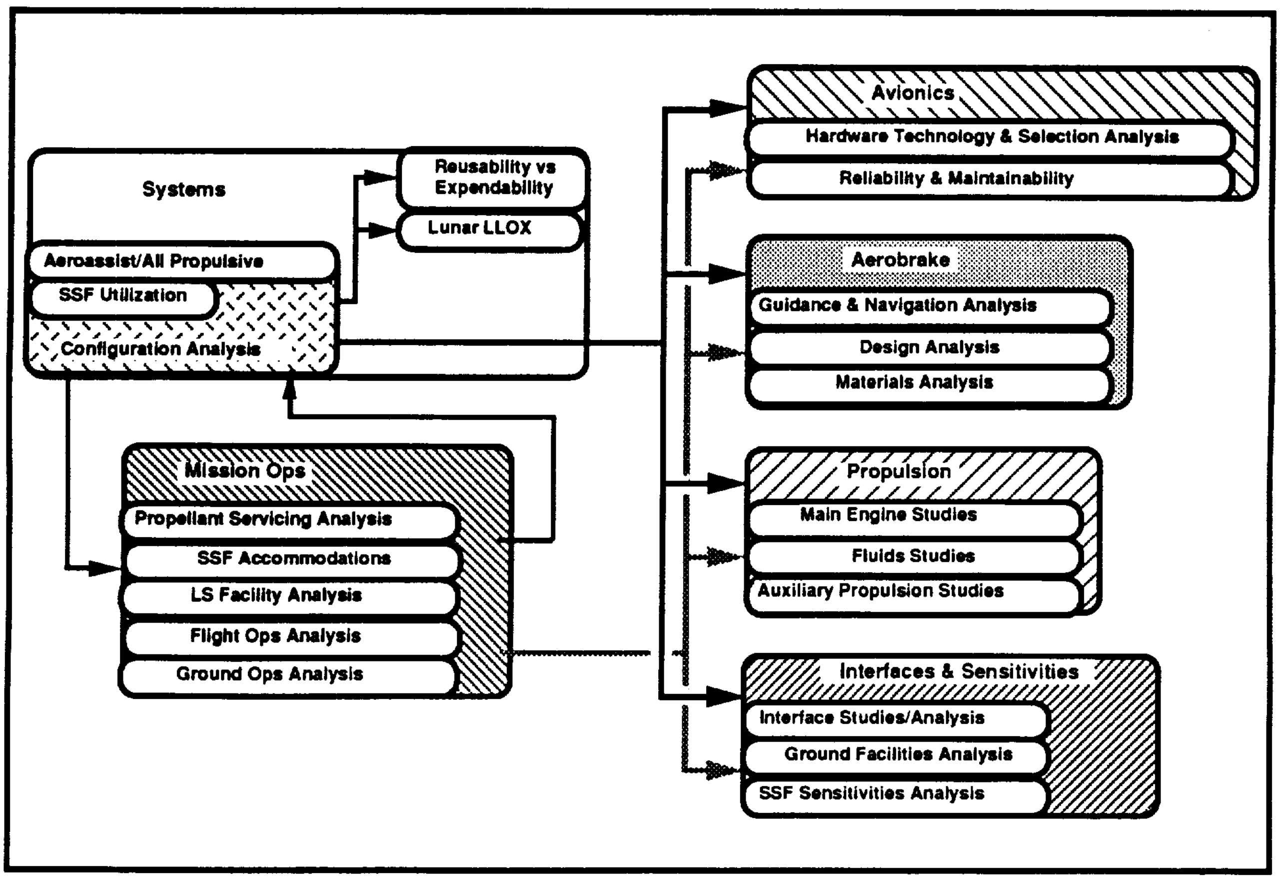


Figure 2.3.1-1 STV Studies & Analyses Approach

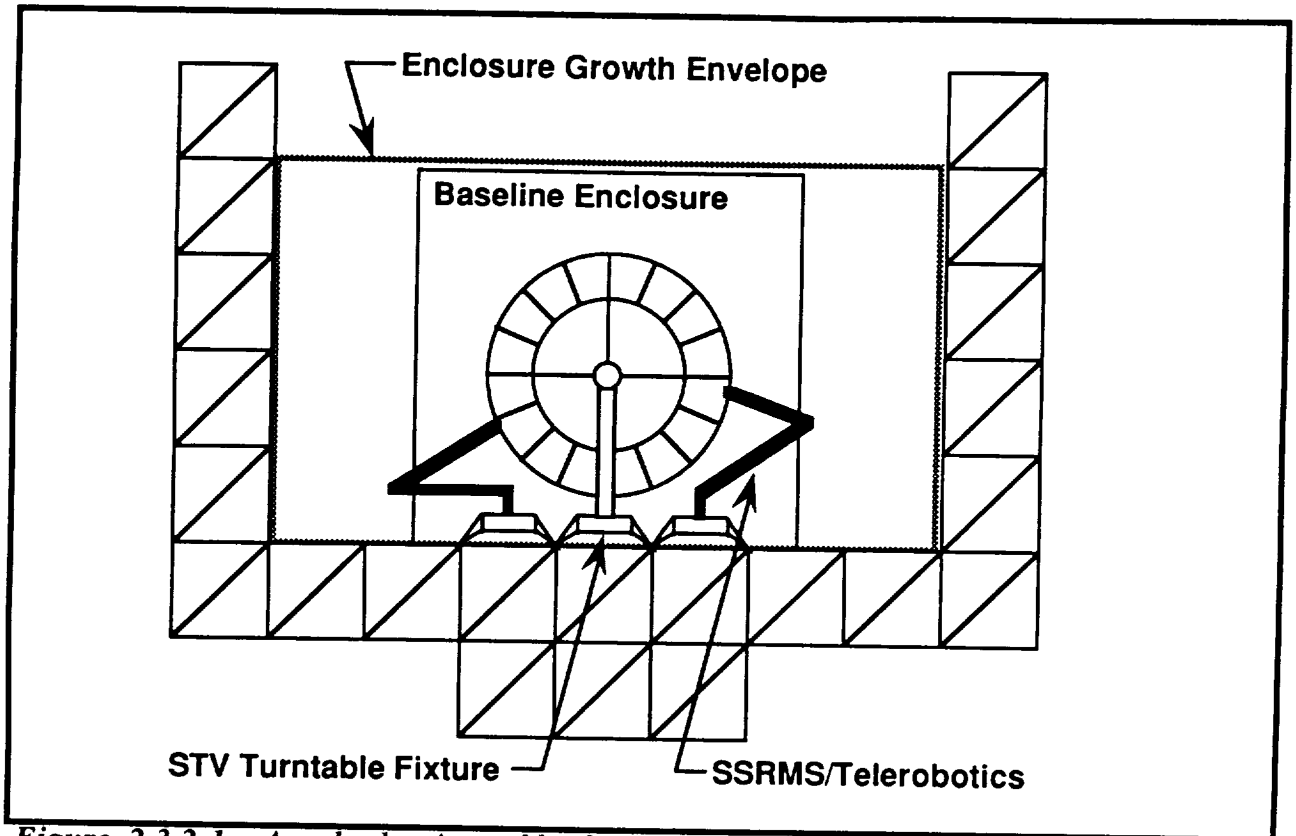


Figure 2.3.2-1 Aerobrake Assembly from within Space Station Freedom STV Servicing Enclosure

Lunar Transportation System (LTS) with an upward and downward evolution to Mars and near Earth missions.

2.3.2.1 Orbital Operations Analysis—Orbital operations analysis assumed the ability of Space Station Freedom to provide support to a space based transportation system. Key areas addressed were the approach to element assembly with an emphasis on the aerobrake and the ability of the SSF crew to provide the necessary support. Primary station sensitivities were not included as part of the operational analysis activity but fall under system level studies and analyses in Section 2.3.3.4.

2.3.2.1.1 Aerobrake Assembly—One of the main SSF based operations for STV servicing was the assembly of the aerobrake. In addition to being intricate, the operational approach will have a large impact on the design of the aerobrake. Three criteria areas; crew resources, task time and technology risk were analyzed for two separate aerobrake assembly operations approaches.

Two primary assembly options were considered for the aerobrake assembly trade study. Both assembly options used the servicing facility equipment and enclosure as shown in Figure 2.3.2.1-1 The assembly techniques, timelines, and impacts were based on the Martin Marietta STV and aerobrake that corresponds to the 90-day study configuration. The use of other STV/aerobrake configurations would modify the resulting timelines and complexities, but would not otherwise substantially alter the conclusions.

Option 1 (IVA/Telerobotic Assembly) involves using the crew inside a Space Station pressurized control center to direct telerobotic operations to assemble, connect, and verify aerobrake assembly. Option 1 utilizes only IVA assisted telerobotics techniques to connect segments, verify connections, close-out TPS, and connect instrumentation. Specialized tools, end effectors, and robotic devices will have to be developed to support TPS work and instrumentation. Also, EVA backup must be provided for the IVA assembly option.

Option 2 uses Extravehicular Activity (EVA) crew to directly assemble, connect and verify aerobrake construction. Use of telerobotics is limited to standard SSF assembly level support. This includes movement and positioning of sections and crewmen only. The TPS close-out and instrumentation connections are completed with hand tools. Option 2 also requires the availability of EVA support devices such as CETA rails, handholds, tether loops, etc. As can be seen in Table 2.3.2.1-1, resource comparisons show equivalent levels of total man-hours to perform the aerobrake assembly, whether accomplished using telerobotics or EVA. However, the use of EVA crewmen imply a substantial operational cost premium over IVA crew usage.

Table 2.3.2.1-1 Aerobrake Assembly Trade Study Results

Option	Man-Hours (EVA/Total)	Serial Task Hours	Technology Risk	Comments
IVA/Telerobotic	0/280.2	140.1	101/150 (High)	Also Requires EVA Dev't
EVA Assisted	125.8/276.3	91.2	97/150 (Med High)	Uses STV turntable

A significant result is the 35% serial task time advantage offered by use of EVA techniques versus telerobotic techniques. If assembly timelines are a pacing item in STV operations, this could prove to be a great benefit.

The approach to technology risk involved identifying the main areas of technology risk, assigning an uncertainty value to each area (which depends on technology area state-of-the-art) along with a criticality value (which shows the potential for schedule impact), and multiplying the values for each area. The higher risk for development of the telerobotics technology on the first option leads to a somewhat higher risk level being assigned. Table 2.3.2.1-2 shows the risk assessment for both the IVA/telerobotic and EVA assisted aerobrake assembly operations.

As a result of studying the aerobrake assembly operations, a set of design recommendations were produced. The significant point involved design of a simply sealing thermal protection system along with positively latching joint mechanisms. If adopted, these recommendations would offer a 28% improvement in assembly time for the telerobotics option, making it comparable to the EVA option.

Table 2.3.2.1-2 Aerobrake Assembly Options Risk Assessment

Option	Technology	Criticality	Uncertainty	Risk	Comments
#1	Segment Joints	5.0	4.0	20.0	
	Strut Design	3.0	4.0	12.0	
	TPS Closeout	5.0	4.0	20.0	Potential flight test failure
	EVA Assembly	3.0	4.0	12.0	Backup for Robotics
	Robotics	5.0	5.0	25.0	
	STV Rotation	3.0	4.0	12.0	
				101.0	
#2	Segment Joints	5.0	4.0	20.0	
	Strut Design	3.0	3.0	9.0	
	TPS Closeout	5.0	4.0	20.0	Potential flight test failure
	EVA Assembly	4.0	5.0	20.0	Backup for Robotics
	Robotics	4.0	4.0	16.0	
	STV Rotation	3.0	4.0	12.0	
				97.0	

Other key design recommendations relate to latches, adjustable struts, alternate TPS closeout, and the STV turntable. Recommendations for the latches include self-alignment and verification, recycle, and positive latching.

Servicing Analysis—It was determined that the proper approach to understanding the impacts and sensitivities of the space station system due to STV servicing operations would be best studied by examining each proposed STV configuration and evaluating the complexity of its individual servicing operations. The methodology followed in evaluating complexity is displayed in Figure 2.3.2.1-2. After developing an exhaustive list of STV servicing tasks, the complexity of each task was described by the factors shown in Figure 2.3.2.1-2. Each configuration was then evaluated as to which subset of the servicing operations were specifically required for each case. Then, the time to perform each task was estimated for each configuration. The final complexity factor for each configuration was produced by multiplying each task complexity by its duration, and summing for all tasks.

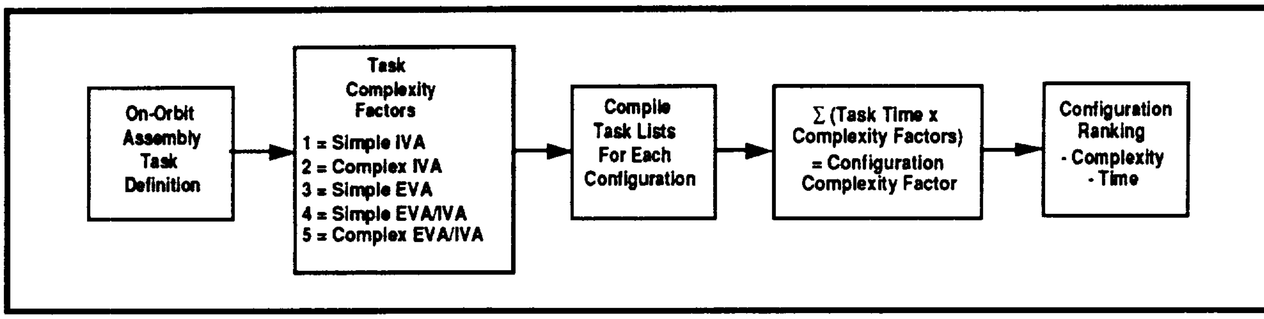


Figure 2.3.2.1-2 Methodology to Determine STV Servicing Sensitivities

In Figure 2.3.2.1-3, the results for each proposed configuration that required Space Station servicing support are displayed. It can be seen that the complexity factor for each cargo and for each crew-carrying STV configuration does not vary significantly. The crew-carrying configurations display much higher complexity factors than the cargo configurations. This is due to the fact that the biggest drivers of the complexity factor were post-flight inspections (of which there are none for cargo configurations) and crew module servicing (which the cargo versions do not have). The complexity factors and crew time estimates were based on a dedicated STV servicing 4-person crew working consecutive two man shifts. For EVA operations, two EVA crewmen would be assisted by a regular space station crewman to monitor operations. If the tasks are not undertaken by specifically trained STV servicing crewmen, then complexity factors could change. This addresses the added issue of additional crew habitation facilities for these special crewmen.

As STV design details are better understood, task complexity factors will change. If, for example, the aerobrake of one configuration is deployable while all others are assembled, the former's complexity factor would be less. Similarly, the locations of engines within the configuration and the servicing requirements for crew module subsystems could be designed to streamline servicing task times, resulting in lower configuration complexity factors.

SSF Crew Size/Utilization Analyses—The SSF crew time analysis that was used as a basis for the candidate tasks and shift times are in the study done by MDSSC-KSC (STV Concept Selection-SS Freedom On-Orbit Operations Evaluations-Preliminary Data-6/2/90 by Don Bryant). The total shift times in the study were multiplied by eight hours and four crew persons to get the total SSF crew hours for each type of mission. For purposes of comparison, 2800 hours was assumed to comprise a SSF man year. See Figure 2.3.2.1-4.

Figure 2.3.2.1-5 contains the same data as in the previous chart with the exception that crew time is converted to a percentage of the available crew time for utilization in the baseline SSF (assembly complete). For the purposes of comparison, an approximate value of 18,000 man hours/year of utilization time was assumed. This was derived from currently hypothesized payload manifest scheduling and utilization operations extrapolated over a year.

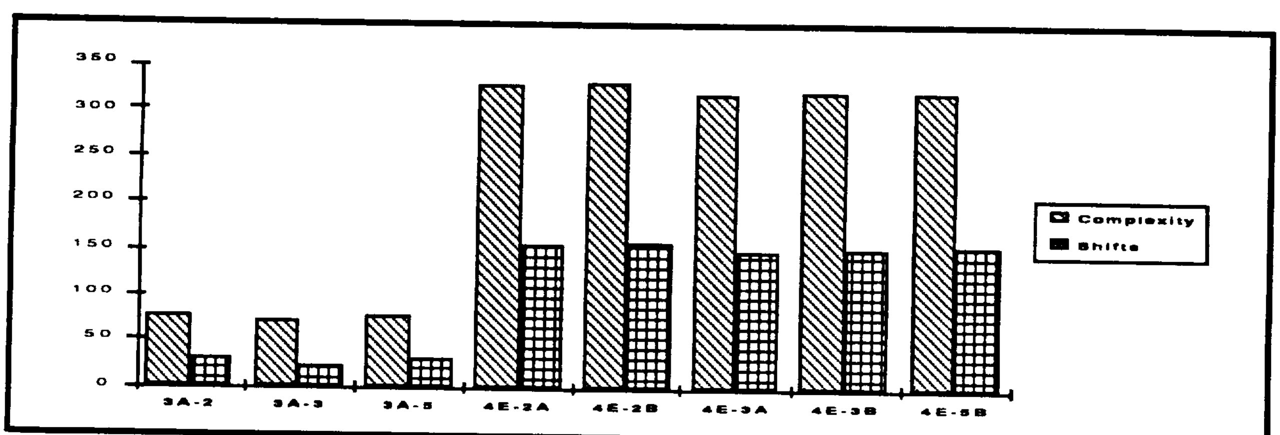


Figure 2.3.2.1-3 Sensitivity of STV Configurations to Servicing Operations

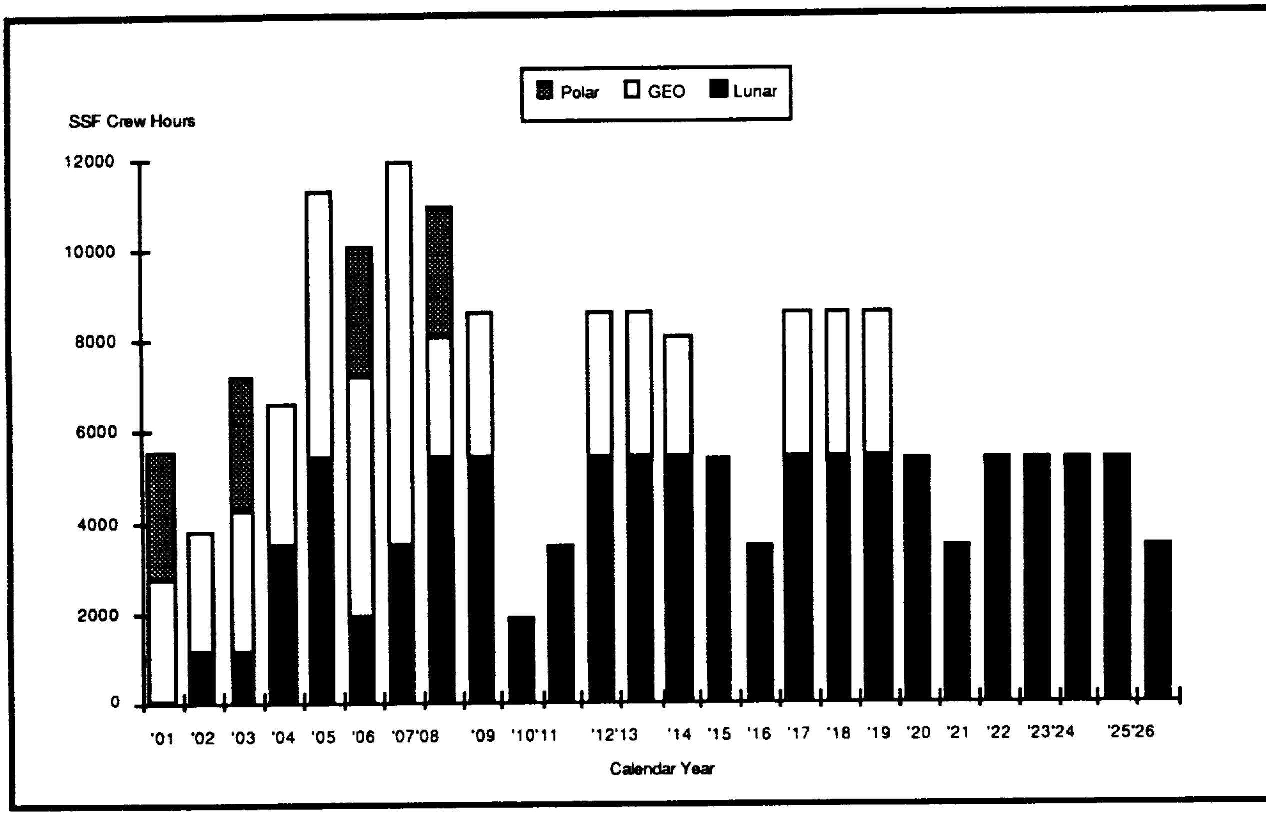


Figure 2.3.2.1-4 SSF Crewtime to Support STV Operations

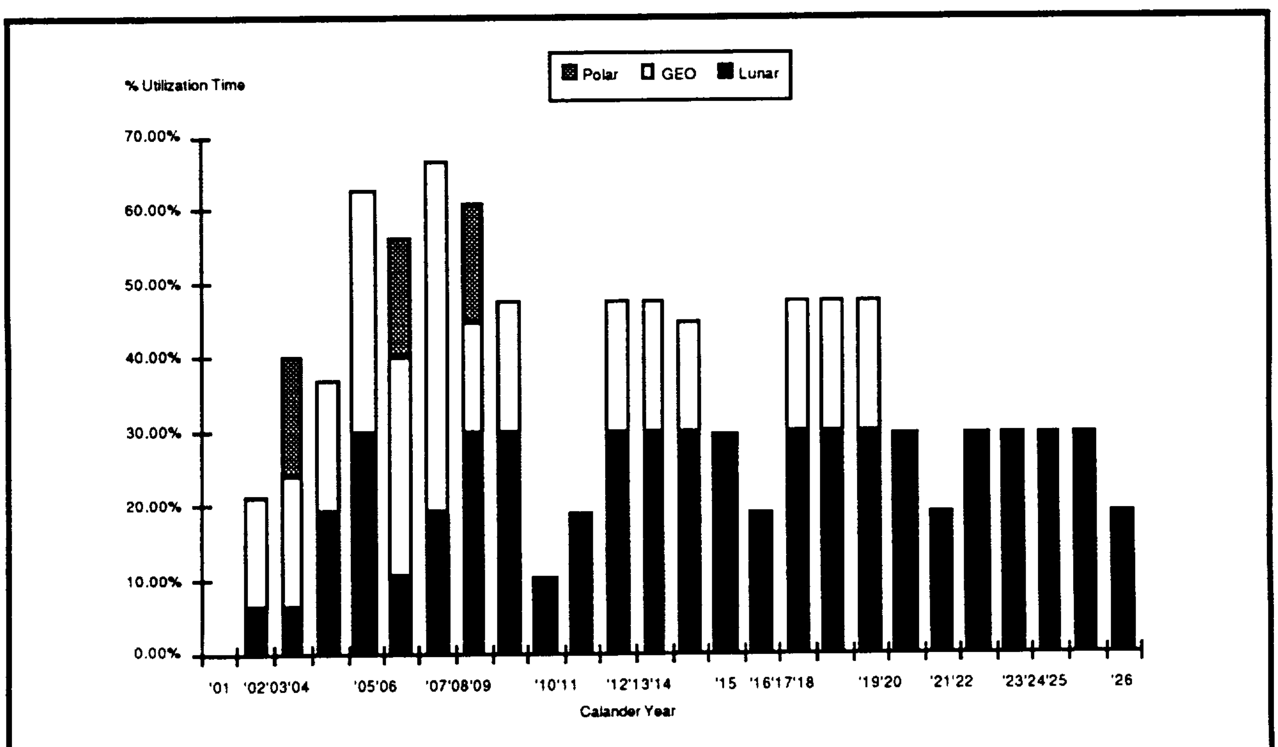


Figure 2.3.2.1-5 % SSF Utilization Time to Support STV Operations

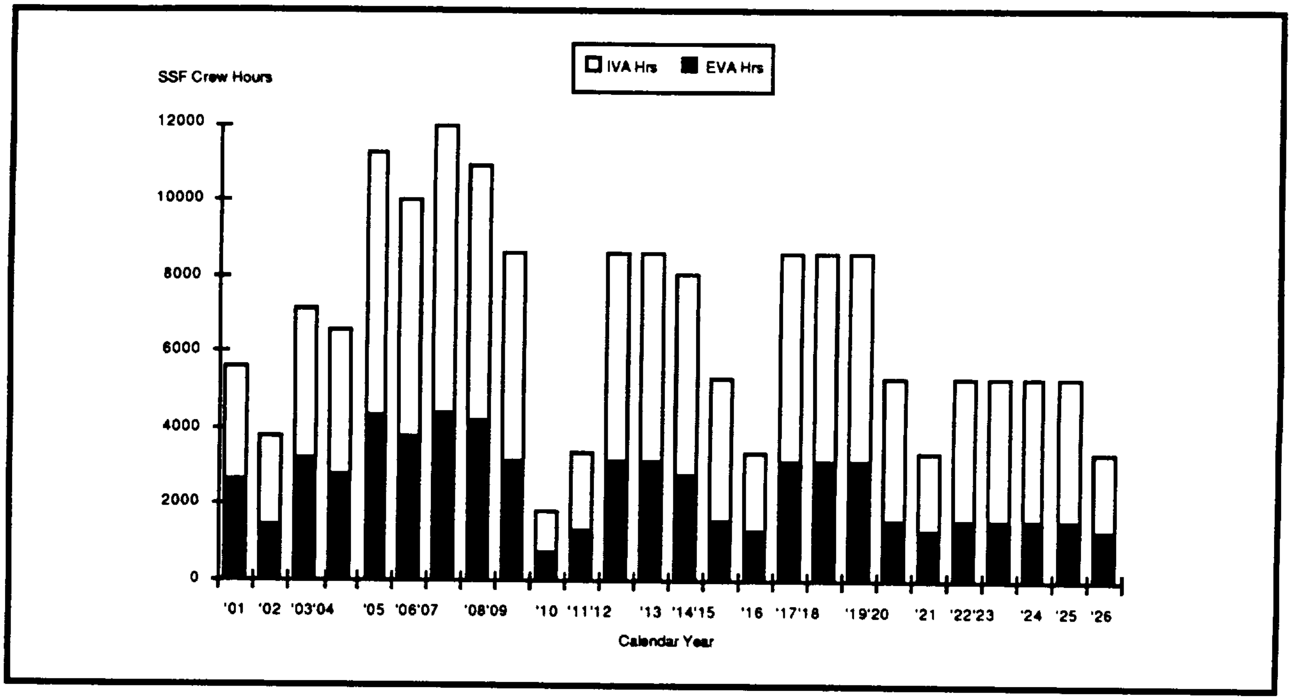


Figure 2.3.2.1-6 EVA / IVA Time Required to Support STV Operations

Tasks requiring EVA operations were assumed to require two EVA crew persons for the duration of the eight hour shift. These values were then multiplied by a factor of 1.5 to account for tasks not directly associated with STV servicing (i.e., crew ingress, attaching/detaching tethers, etc.).

2.3.2.2 Flight Operations Analysis—The flight operations analysis has been separated into two areas. The primary area of activity involved analysis of lunar missions including trajectories, aeroassist maneuvers, and mission times. The secondary area of analysis addressed a ground-based approach involving a high energy stage in support of meeting the STV DRMs.

Lunar Mission—This analysis section addresses the development of the baseline mission architecture and the corresponding Earth-lunar trajectory, mission performance for manned and unmanned configurations, a strategy for free return, Earth-lunar transfer times, and preliminary feasibility of the HLLV upper stage.

In support of the concept selection process, several assumptions were made to enable calculation of the propellant requirements. Table 2.3.2.2-1 lists the sizing groundrules and assumptions used to determine the required propellant loads. All weights refer to Earth surface weights.

Table 2.3.2.2-1 Groundrules and Assumptions

Tank Fraction	4% of Propellant
Leg Fraction	2% of Landed Mass
Structure	2% of Gross Vehicle Mass (no P/L)
Aerobrake	20% of Vehicle Gross at Aeroentry
Engine T/W	30
Vehicle T/W	
Earth Escape	0.25
Lunar Surface	0.5
2nd Stage TV	0.1
Flight Performance Reserve	2% by Velocity
Unusable Propellant	1.56% of Total Propellant
Avionics	0 (In the noise)

- TV-Crew Module Mass, Including 4 crew, suits and consumables: 9760
- LV-Crew Module Mass, Including 4 crew, suits and consumables: 3130
- Single Stage combined Vehicle Expends the Following on the lunar surface: Structure mass and Leg mass
- Multi-Stage vehicles driven to common size
- Drop Tanks always dropped after TLI
- Drop tanks sized for Entire Propellant load
- Engine Performance Based on RL-10B-2 (Isp = 460 sec)

Four percent of the propellant weight was the figure used for the weights of the main propellant tanks. Landing vehicle legs weighed two percent of the total vehicle weight at the time of touchdown. Vehicle structure for transfer vehicles (TVs) and landing vehicles (LVs) was two percent of the total gross weight. Aerobrakes were assumed to be 20 percent of the total vehicle weight at the point of atmospheric entry. Engines were "rubber" and sizing was based on the required vehicle thrust-to-weight. Engines weighed one thirtieth of the thrust they generated, which, for example, is one quarter of the weight of a TV in an Earth escape burn. Flight performance reserve was done by velocity, not propellant. Propellant requirements, per maneuver, were sized to 1.02 times the Δ-velocities found on the following chart. Unusable propellant was assumed to be 1.56 percent of the propellant loaded. In the case of drop tanks, the corresponding unusable propellants were jettisoned with each tank. For core vehicle tanks, this propellant is effectively stuck in the core and subsequently increases its inert weight. The transfer vehicle's crew cab, including four suited crew members and consumables has a mass of 9760 kilograms and the landing vehicle's crew cab has a mass of 3130 kilograms, including suited crew. Single stage combined vehicles left the legs and some excess structure mass on the lunar surface when an Earth return was required. For cost savings, if more than one TV is used, the sizing routine was driven to match sizes of the TVs so that several unique stage designs could be avoided. In the early phases of the concept selection process, the assumed Isp for all engines was 460 seconds, which was a compromise between existing engines (RL10A-4, ~450 sec) and the advanced space engine (ASE, ~475 sec). However, as the selection process progressed, the Isp was increased to 476 seconds for space transfer vehicles and was held at 460 seconds for lunar landing vehicles.

In support of the configuration analysis task, the five primary mission architectures shown in Figure 2.3.2.2-1 were defined and analyzed, and a recommendation was made as to which architecture should be used as the baseline for both performance and operations analyses. Those five architectures included LEO transportation node (baseline), LEO crew node, No LEO transportation node, LEO crew return node, and LEO crew node/Earth return. An evaluation of these architectures and their supporting vehicle concepts indicated that the LEO transportation node was the best relative architecture. This was due to many factors - cost, risk, operations, and mission adaptability. It should be recognized that this decision is dependent on the assumptions that were made, as well as the relative weighting of the various selection criteria.

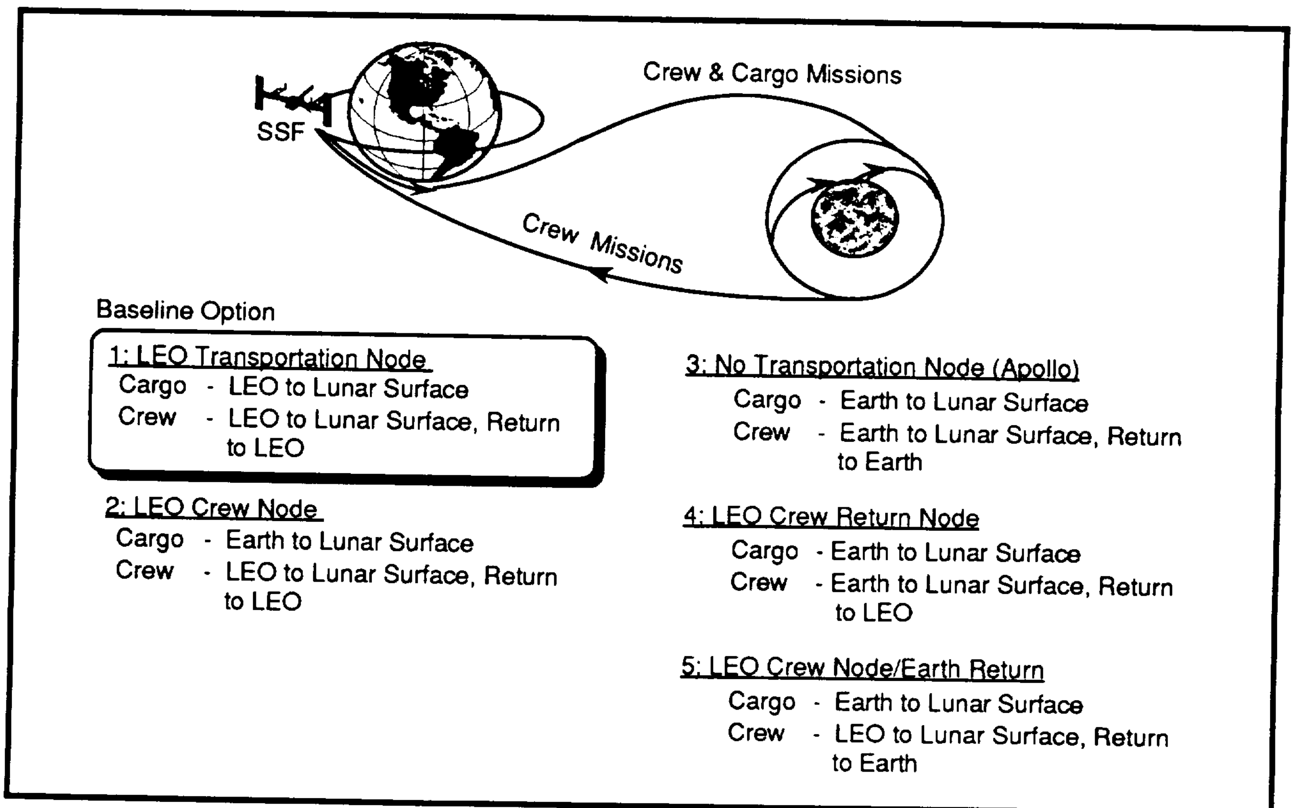


Figure 2.3.2.21 Lunar Mission Architectures

Based on the selected architecture analysis, an Earth-lunar trajectory was defined (Fig. 2.3.2.2-2). The major burns are accomplished using the main engines, while the Trajectory Correction Maneuvers (TCMs) are accomplished using the reaction control system. The lunar descent (not shown) is initiated after LOI and separation from the aerobrake element occur. Following the lunar stay, the vehicle ascends (not shown) to LLO and rendezvous with the descent aerobrake element (not shown) before its return home. Both the descent and ascent phases of the mission are accomplished using the main engines. The baseline Earth orbit altitude is 407 kilometer (SSF orbit), and the baseline lunar orbit altitude is 300 kilometer. It should be noted that the Δ-velocities shown for LOI and TEI include an allowance for a five degree inclination change while in LLO (Moon at apoapsis). If the Moon is at the periapsis of its orbit, this allowance can be increased to as much as eight degrees.

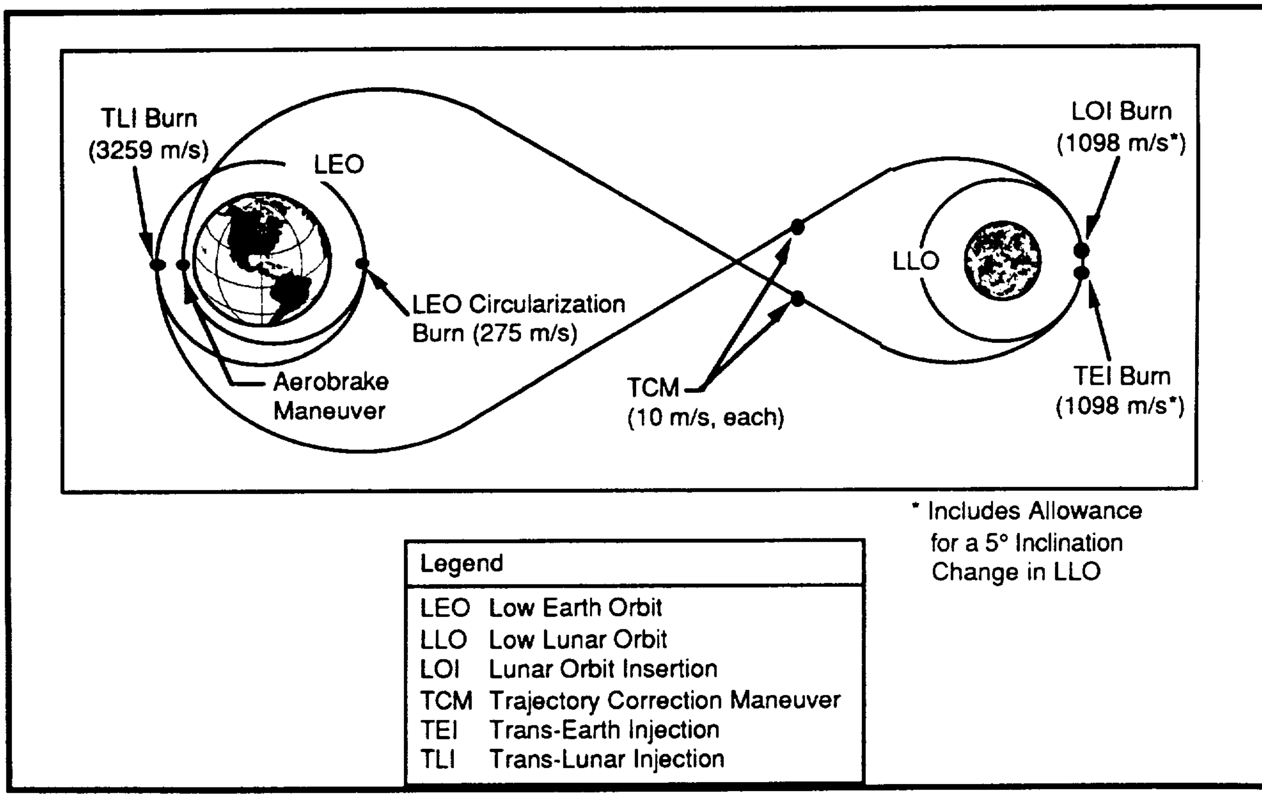


Figure 2.3.2.2-2 Baseline Earth-Lunar Trajectory

To supplement this trajectory analysis, a strategy was developed that would allow for the return of the vehicle and crew to SSF without the use of a seperate vehicle (Fig. 2.3.2.2-3). Since a direct free return to SSF is generally not possible due to the plane of the vehicle on Earth return not aligning with SSF's orbital plane, two steps are used to achieve the recovery of the vehicle at SSF.

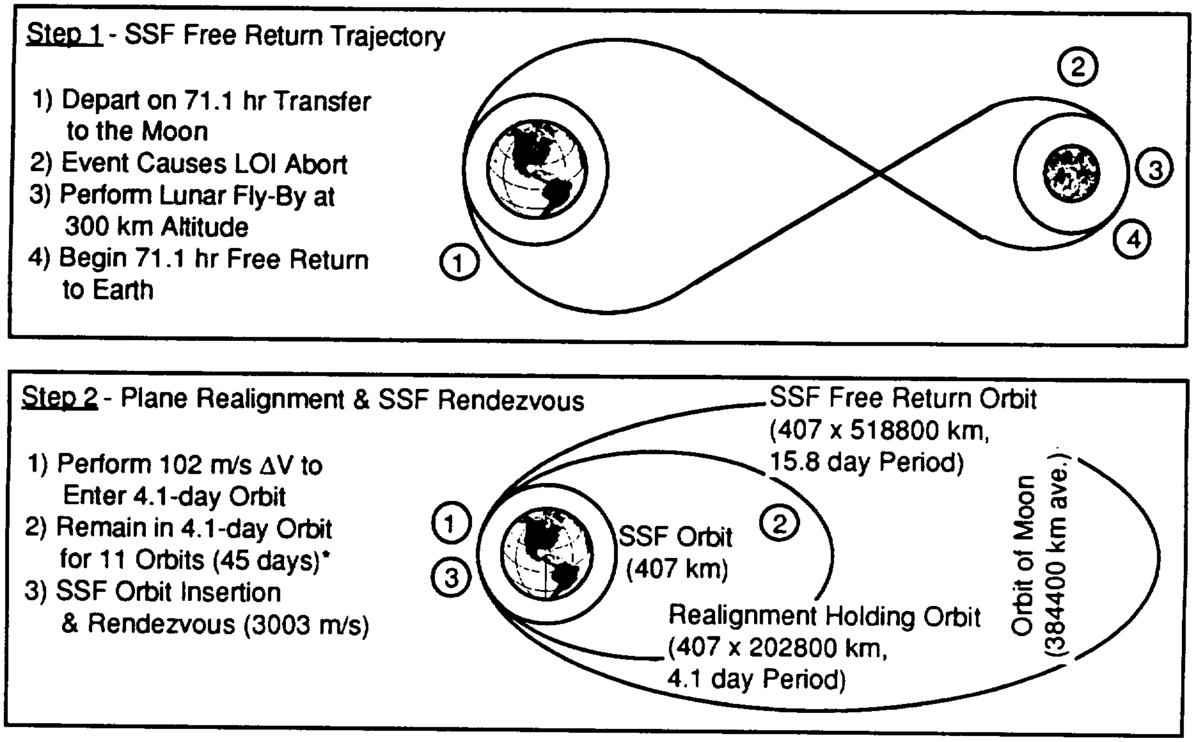


Figure 2.3.2.2-3 Two Step SSF Free Return Strategy

To supplement this trajectory analysis, a strategy was developed that would allow for the return of the vehicle and crew to SSF without the use of a seperate vehicle (Fig. 2.3.2.2-3). Since a direct free return to SSF is generally not possible due to the plane of the vehicle on Earth return not aligning with SSF's orbital plane, two steps are used to achieve the recovery of the vehicle at SSF. The initial step begins with the vehicle departing on a 71.1 hour free return trajectory to the Moon, with a lunar fly-by altitude of 300 kilometer. Once the decision has been made to execute the free return, the vehicle would perform the 300 km lunar fly-by and embark on the 71.1 hour return to Earth. Once at Earth, the vehicle would begin the second step, performing a 102 meter/second retro-burn at periapsis to change the vehicle's orbit from a 407 x 518814 kilometer, 15.8-day orbit to a 407 x 202800 kilometer, 4.1-day orbit. The vehicle would then remain in that holding orbit for 11 complete orbits (~45 days), allowing SSF's orbit to precess into the plane of the elliptical orbit. After the orbital planes are realigned, the vehicle would make the final 3003 meter/second retro-burn to insert into SSF's orbit and then rendezvous with SSF. Our baseline vehicle would employ its aerobrake to achieve both the 102 meter/second and 3003 meter/second Δ-velocities if its main propulsion had failed. Because the vehicle would pass through the Van Allen radiation belts several times while waiting for SSF rendezvous, it might seem that the crew would be exposed to an inordinate amount of radiation. However, a separate study has determined that the crew's exposure to radiation while in a 4-day orbit is actually less than it would be for the same amount of time spent in LEO.

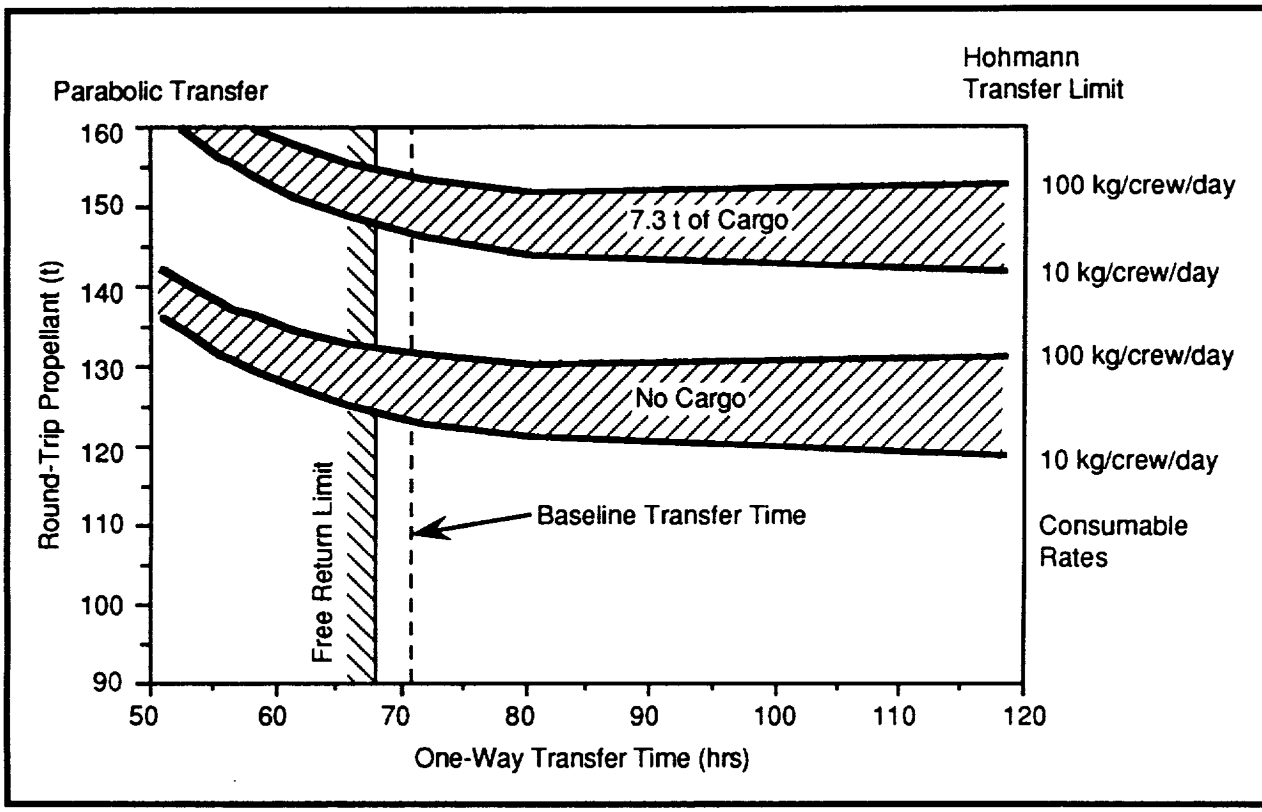


Figure 2.3.2.2-4 Total Round-Trip Propellant versus Transfer Time

Once the baseline mission architecture and trajectory were defined, a detailed analysis was conducted to optimize the effect of one-way transfer time on the total propellant load, assuming that both legs of the round-trip mission had the same one-way time. Figure 2.3.2.2-4 shows total round-trip propellant as a function of one-way trip time for cargo loads of 7.3 and 0.0 tonnes. The graph also shows the effects of the crew's consumable rates on the round-trip propellant. A free return trajectory with a lunar fly-by altitude of 300 km would have a one-way transfer time of ~71 hours, with transfer time increasing (up to ~120 hours) with increasing lunar fly-by altitude. The minimum one-way transfer time for a free return is ~68 hours (0 km lunar fly-by). The left border on the graph represents a parabolic Earth departure and is not a physical boundary, i.e., hyperbolic Earth departures and lunar orbital captures are possible. However, the right border on the graph is a physical boundary and represents the lowest energy elliptical transfer possible.

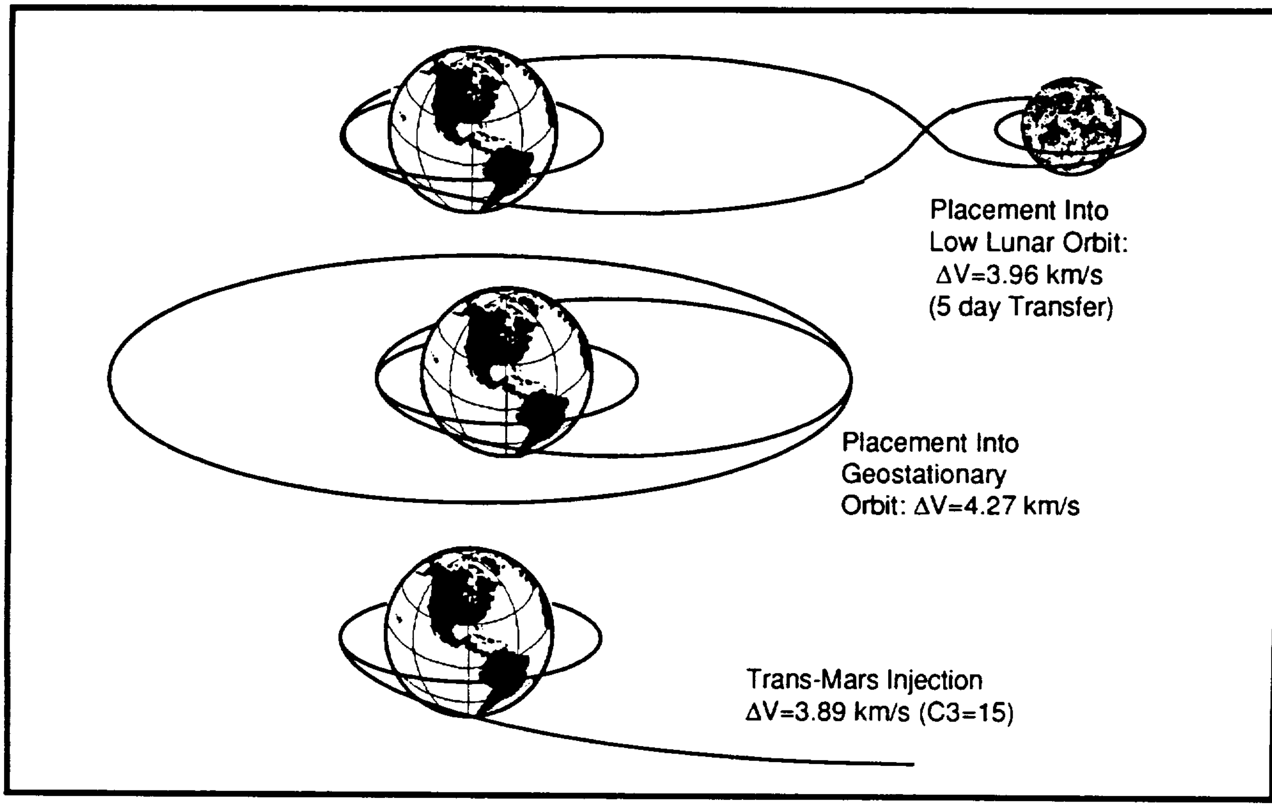


Figure 2.3.2.2-5 Similar Δ-Velocity Mission Requirements

2.3.2.2.2 Ground Based Missions—Since the baseline STV presented in the rest of this document is dominated by requirements that came from the 1989 90-Day Report (Skunk Works), it is important to assess what requirements could be generated without the emphasis on space-basing and reusability. Figure 2.3.2.2-5 shows how three important mission classes all require about 4 km/sec Δ-velocity from LEO. It requires 4 km/sec to put a satellite into geostationary orbit (GSO) and into a trans-Mars trajectory (actually many different energy levels can be used, but for this example a C_3 of 15 km^2/sec^2 is assumed). Finally, the lunar mission requires about 3.1 km/sec to send a payload toward the Moon; however, since most payloads are destined for the lunar surface, placement into LLO is an equally viable mission for the STV. From LLO a lander vehicle can then take the payload to the surface. When adding the 860 meter/second Δ-velocity to the 3.1 km/sec Δ-velocity, the total Δ-velocity is 3.96 km/sec. Hence, for the commercial GSO market and the two objectives of SEI - the Moon and Mars - we see that a common stage can be designed.

Prior to conducting the in-depth analysis required to substantiate a high energy upper stage approach, a set of groundrules and assumptions beyond those defined for the overall analysis activities was developed. Table 2.3.2.2-2 defines the unique groundrules and assumptions that were used in the study of an upper stage for a shuttle derived HLLV. The study assumed the use of a circular park orbit at 185 kilometers and 28.5 degrees. This park orbit was used because most high energy missions use LEO to minimize their total mission Δ-velocity by selecting the optimum time to start the transfer burn, i.e., nodal crossing. LEO is also used for final targeting and improves mission flexibility by increasing the width of the ETO launch window. In all cases, the booster vehicle consisted of two Advanced Solid Rocket Motors (ASRMs), an External Tank (ET) derived core, and a payload shroud based on our Advanced Launch System work. The differences lie in the type and number of engines used and the manner in which they were mounted on the core. The two engines considered were the Space Shuttle Main Engine (SSME) and the Space Transportation Main Engine (STME). These engines were used in sets of three and four and were mounted in either a side-mount or in-line fashion. The characteristics for each of these engines are shown. The upper stages were sized parametrically, but all were based on the assumptions listed on the chart. The upper stages had thrust levels ranging from 444 kilonewtons (100 kilopounds) to 1332 kilonewtons (300 kilopounds) and propellant loads ranging from 45 tonnes (100 kilopounds) to 160 tonnes (350 kilopounds).

Table 2.3.2.2-2 Upper Stage Groundrules and Assumptions

Park Orbit: 185 km x 185 km, @ 28.5°

Booster	
2 x ASRMs Total Mass	1214.5 t
External Tank Usable Propellant	723.4 t
External Tank Inert Mass	35.6 t
Engine & Support Structure Mass	Variable
Payload Shroud Mass	20.4 t
SSME Vac Thrust (@ 104%)	2171 kN
SSME Vac Isp	453 sec
STME Vac Thrust	2576 kN
STME Vac Isp	439 sec

Upper Stage	
Thrust	Variable
Isp	470 sec
Engine Thrust-to-Weight	50
Tank Fraction*	0.035
Structure Fraction*	0.02
RCS Tank &Propellant Fraction*	0.01
Helium Tank Fraction*	0.005
Unusable Propellant*	0.01

* Fractions of Total Propellant

The performance advantages that this stage offers are shown in Figure 2.3.2.2-6. By going to three ASRMs and extending the length of the ET, the 1.5 stage HLLV has been sized to match the LEO capability of one of the eight 2.5 stage vehicles evaluated. But as the Δ-velocity increases, the capability of the 1.5 stage HLLV falls off much more rapidly than does the capability of the 2.5 stage vehicle. For example, the Geostationary Transfer Orbit (GTO) capability of the 2.5 stage vehicle is roughly twice that of the 1.5 stage HLLV. Furthermore, at 4 km/s the 1.5 stage HLLV's capability drops to zero while the 2.5 stage vehicle gets ~45 tonnes. The three missions previously mentioned as having a Δ-velocity of approximately 4 km/s have been highlighted.

Analysis of the 4 km/sec stage was conducted over a range of potential HLLV systems since the exact configuration and capabilities of the HLLV have not been formulated. Figure 2.3.2.2-7 represents an HLLV & upper stage system that was optimized for the 4 km/sec mission. Based on

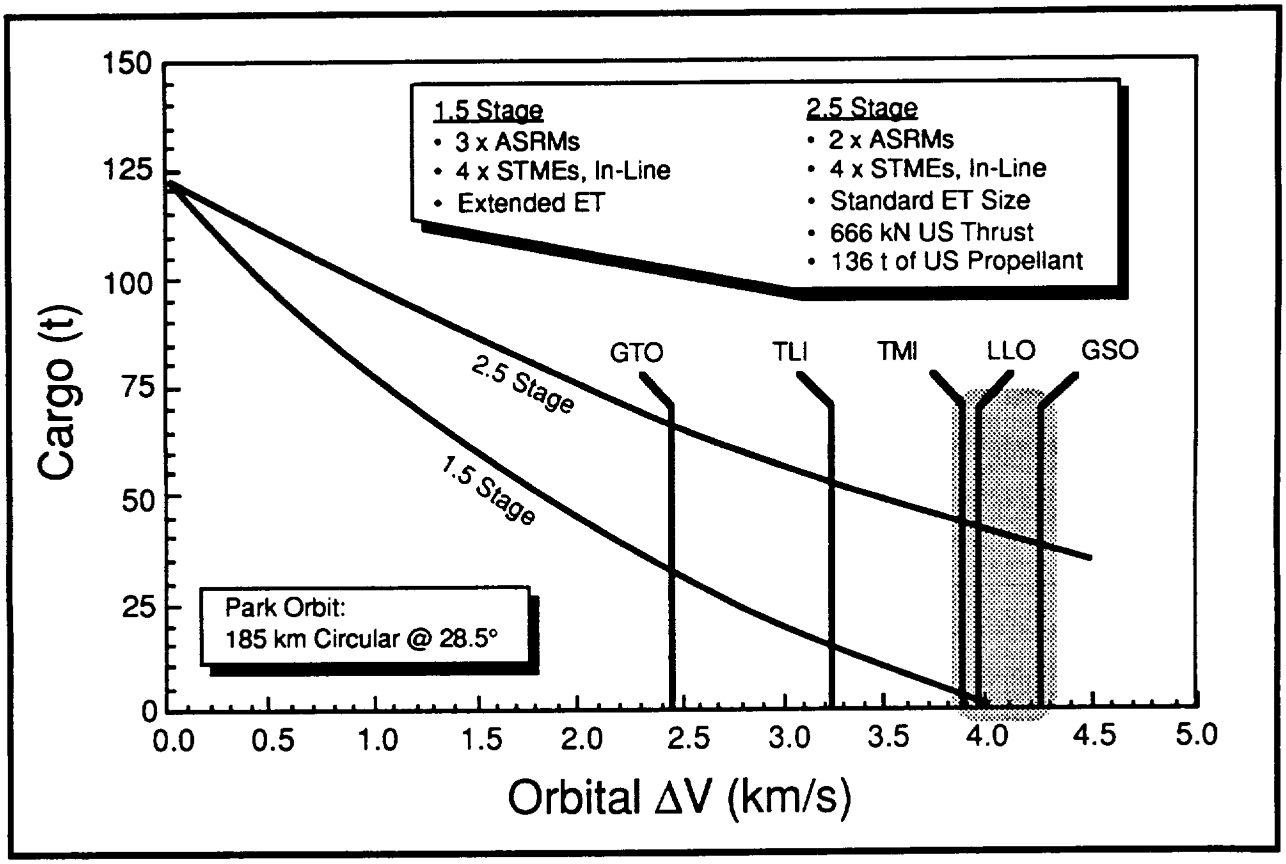


Figure 2.3.2.2-6 Cargo vs. Orbital Δ-Velocity With/Without an Upper Stage

a shuttle derived HLLV which has 2 ASRMs, an ET derived core (with the standard propellant capacity) and four SSMEs mounted in-line, this configuration can deliver a maximum of 44.5 tonnes to a speed of 4 km/sec beyond LEO speed. To attain the maximum 44.5 tonnes, the propellant load in the STV and the thrust level were varied until an optimum was achieved. Hence, rough requirements for these parameters are presented as 136.1 tonnes of usable propellant and 666 kilonewtons of thrust assuming a specific impulse of the upper stage's engine of 470 seconds. This thrust level and specific impulse could be achieved in many ways, i.e., ring of RL10A-4s, ring of RL10B-2s, platelet engines, or ASEs. This configuration can land 19.5 tonnes on the lunar surface with a storable propelled lander in an expendable mode. As stated before, it can also send slightly more than 44.5 tonnes toward Mars or place slightly less into GSO. Figure 2.3.2.2-8summarizes the performance data representing cargo as a function of thrust level for all eight HLLV configurations considered. It assumes that for any given thrust level the optimum propellant load is being carried. Some of the curves have valleys in them due to the dependency of cargo capability on the thrust of the upper stage, the inert weight of the upper stage engines, and the HLLV booster configuration.

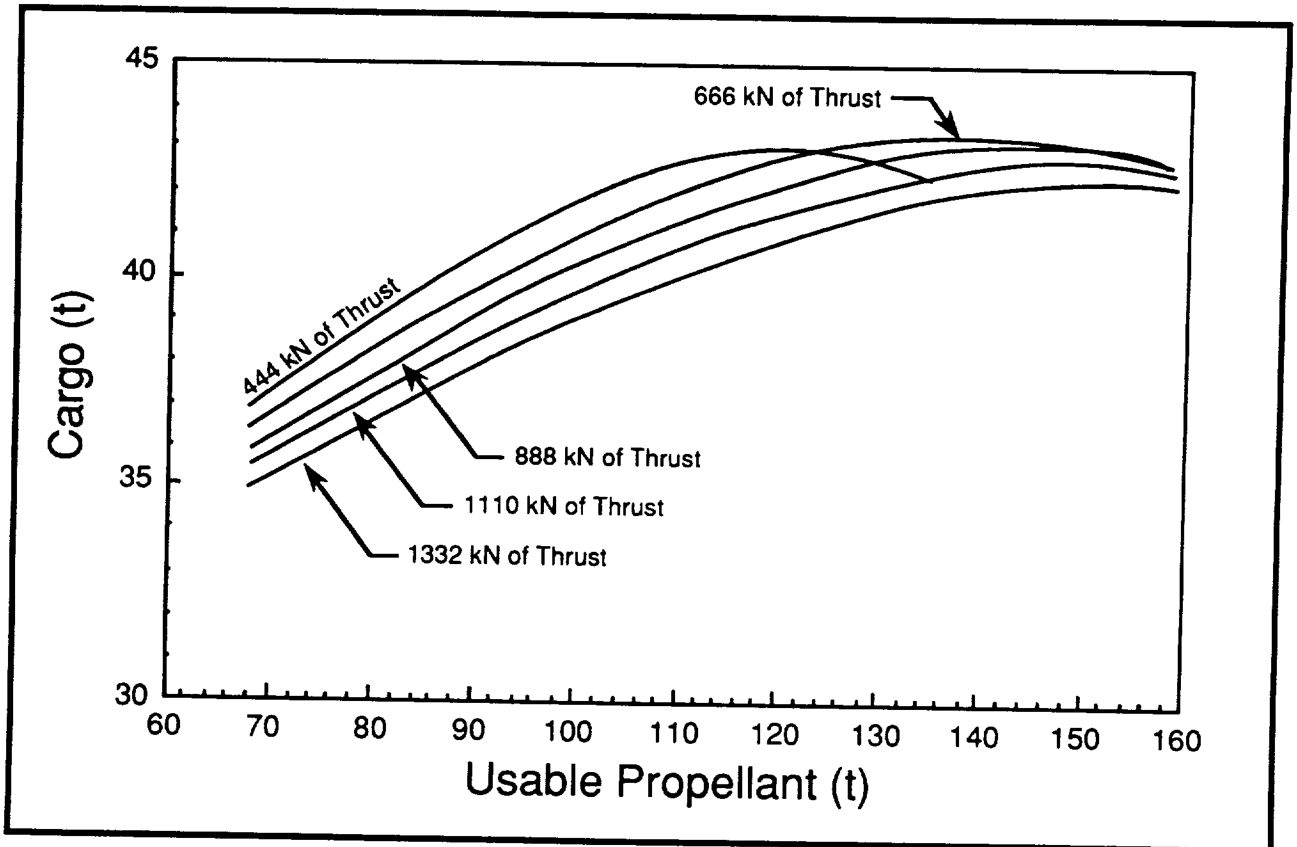


Figure 2.3.2.2-7 km/s Upper Stage - HLLV with 4 Side-Mounted SSMEs

2.3.3 Systems Analysis

Following the definition of the STV requirements base and in conjunction with the mission analysis effort, four major system studies were conducted. These studies included propulsion, basing, aeroassist, and design. The propulsion study traded the use of chemical propulsion against identified advanced propulsion techniques such as nuclear thermal and electric. Although the results of this trade indicated that there were economic and performance benefits associated with the advanced systems, the STV contract SOW dictated the use of chemical propulsion systems in

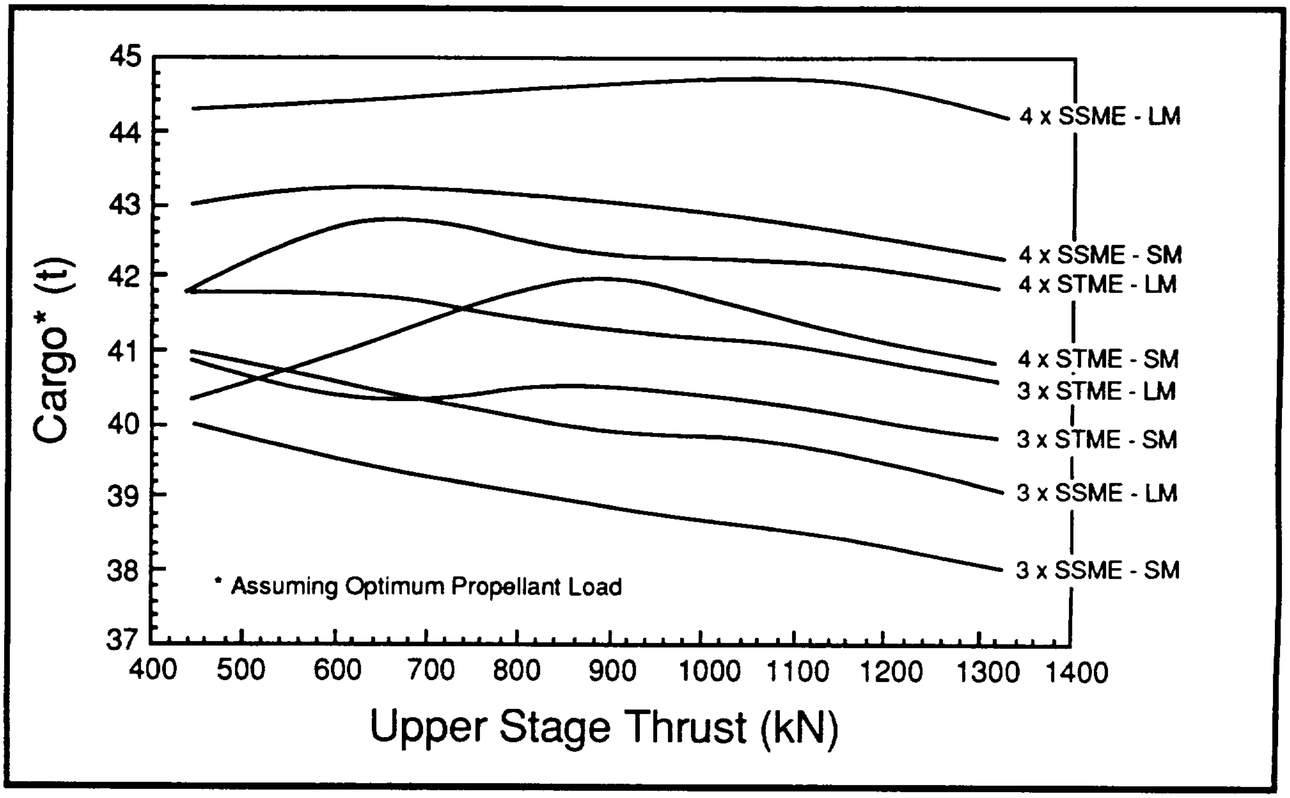


Figure 2.3.2.2-8 Upper Stage Cargo versus Thrust

the analysis of the system configurations. The aeroassist study addressed the feasibility of using an aerobrake for earth return versus use of an all-propulsive system. The results of this effort indicated that until the aerobrake mass fraction reaches 50% the aerobrake provides a lower IMLEO requirement. Data from the mission, engine, and aeroassist studies are used as baselines to conduct the largest of the systems studies, the configuration selection analysis. Within this analysis the implementation of man-rating on the transportation system was evaluated along with the systems programmatics that included test, cost, and schedules.

2.3.3.1 Aeroassist vs All-propulsive Analysis —The objective of the aeroassist versus all-propulsive study was to determine relative LCC benefits as a function of the aerobrake mass fraction, ETO specific costs ($/mass), and the costs associated with development of the aerobrake. The study showed that even if greater aerobrake mass fractions are required than currently estimated (11% to 15%), the LCC benefits are still substantial, see Figure 2.3.3.1-1.

One of the more critical elements in establishing aerobrake and total system development cost is the question of the need for subscale flight testing. Preliminary studies have shown that flight testing an approximately half scale prototype aerobrake could be accomplished using the existing STS as the launch vehicle. However, such a test or tests would add significantly to the cost of aerobrake development. Further assessment of the pros and cons of such testing is required. Relative to the issue of aerobrake reusability, the LCC cost study results suggest that, depending on development costs, the cost advantage the aerobrake affords should not disappear even it is only used one

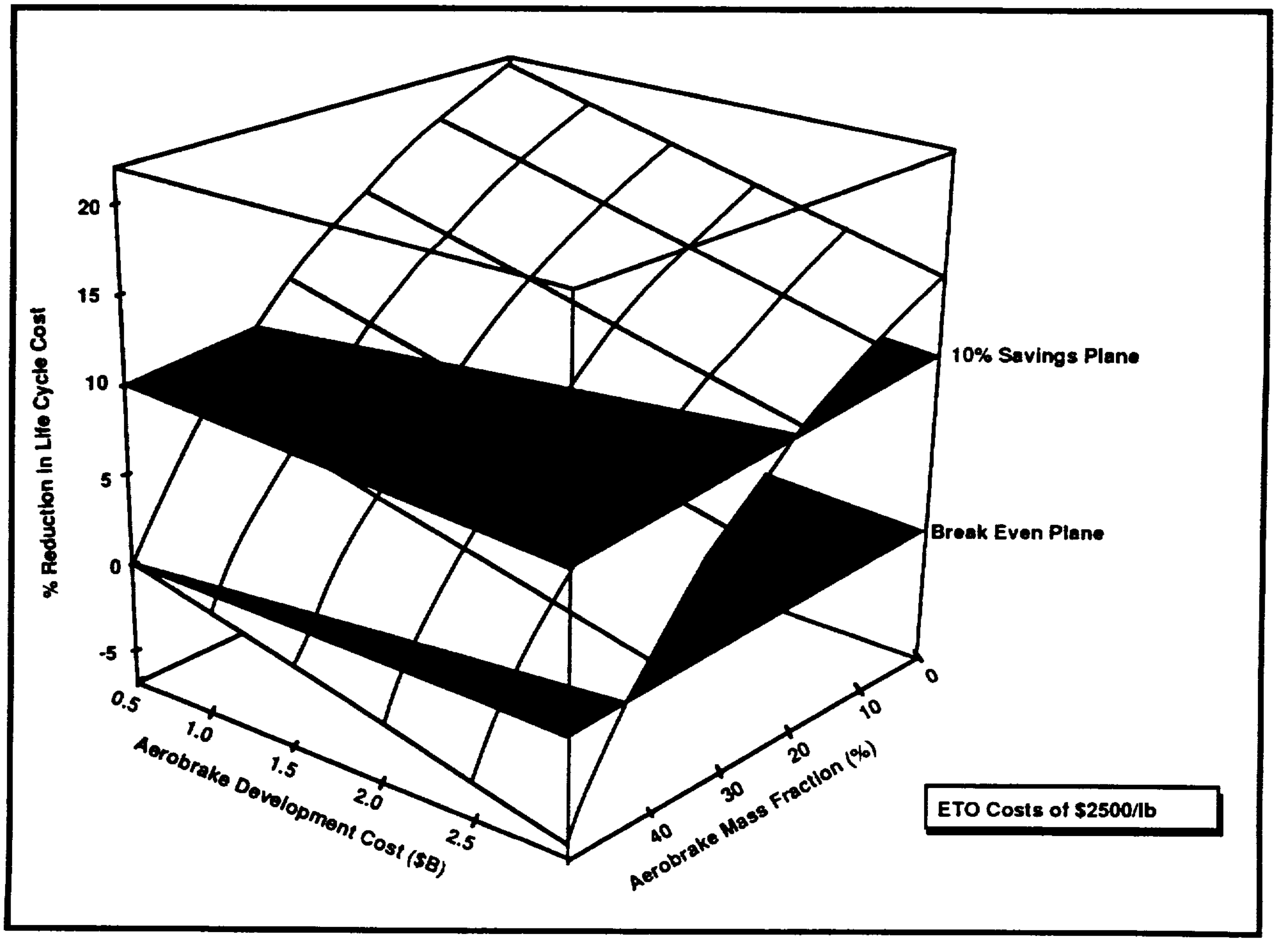


Figure 2.3.3.1-1 Aerobrake LCC Saving Relative to All Propulsive

time. (Complications in ETO manifesting associated with replacement of the aerobrake more frequently than other subsystems have not been evaluated). Another concern, afterbody heat protection during the aerobrake maneuver, also has not been evaluated sufficiently due to wake heating uncertainties. There appears to be room to increase system mass for this purpose without significantly eroding the cost advantages of the aerobrake approach, although adding heat protection to the core vehicle has a two to three times greater impact on IMLEO mass as does adding mass to the aerobrake since the core vehicle descends to the lunar surface.

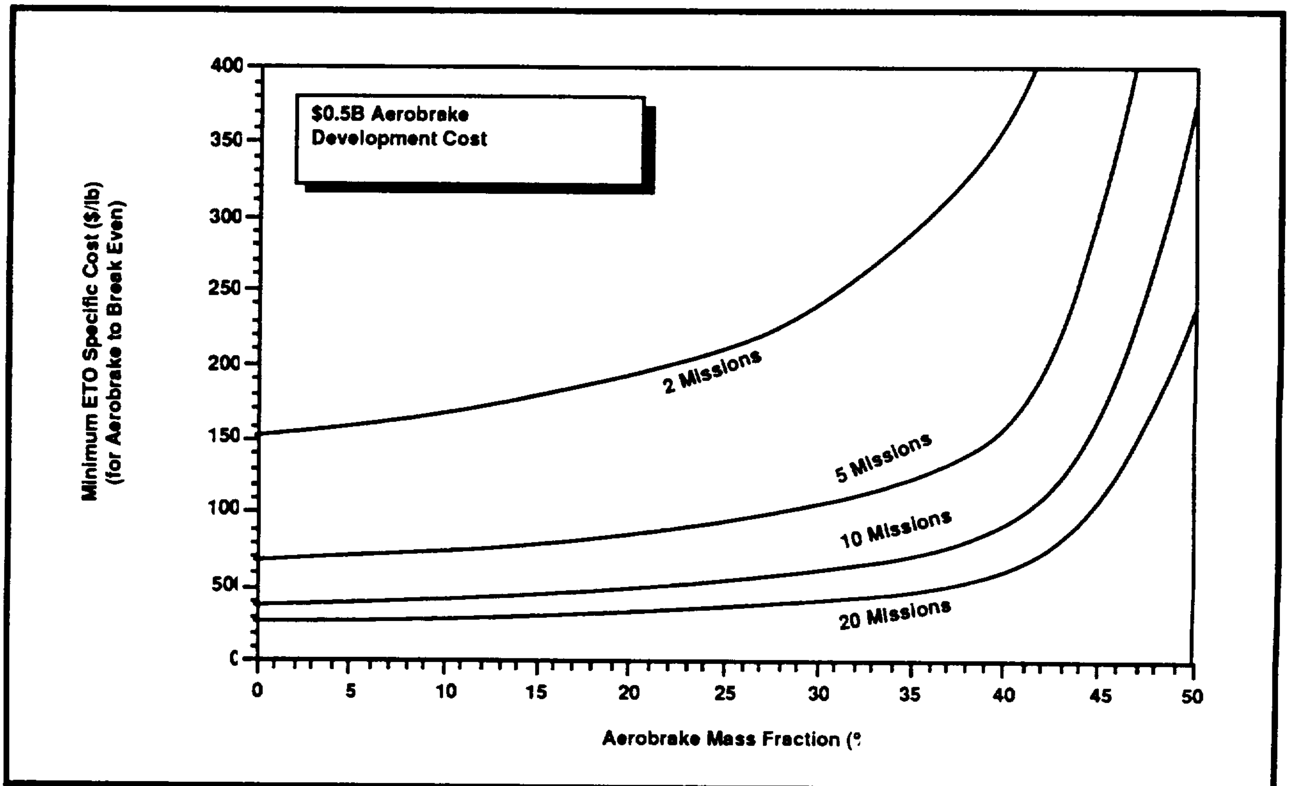


Figure 2.3.3.1-2 Trends in Number of Mission Required to Amortize Aerobrake Development Costs

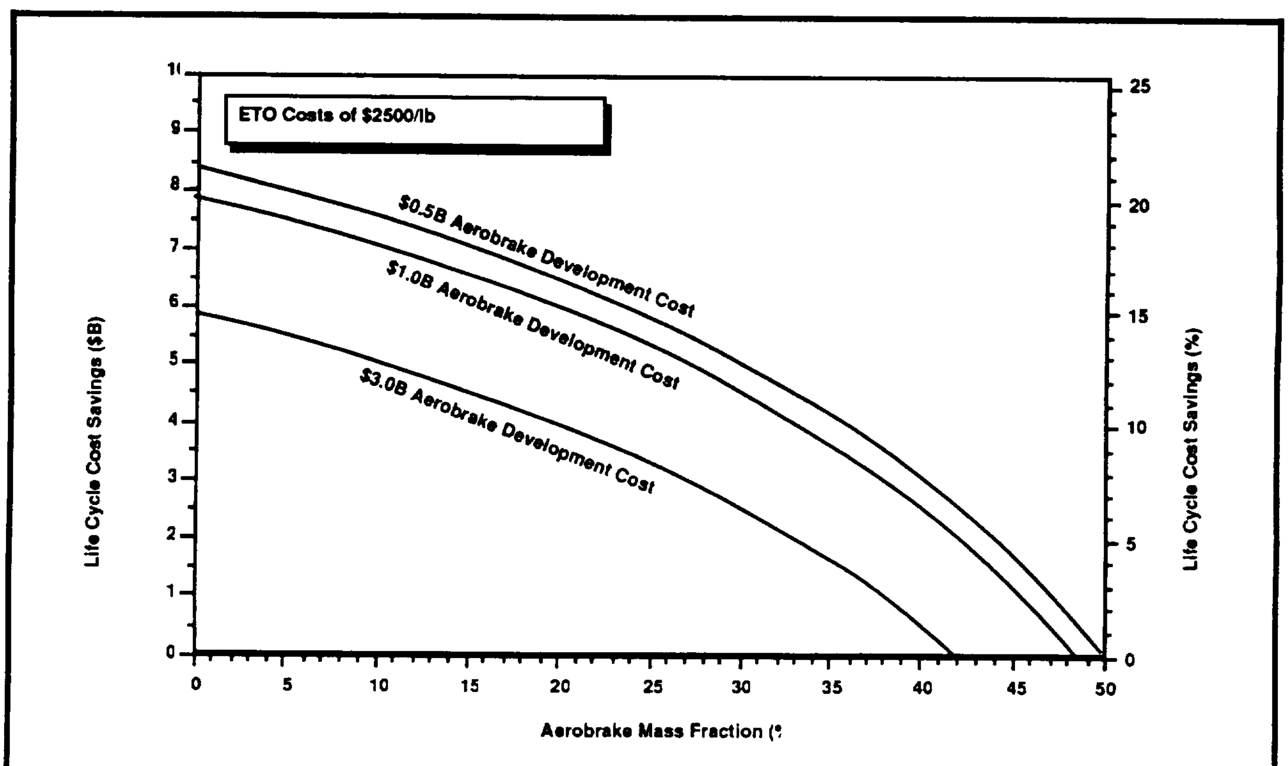


Figure 2.3.3.1-3 Cost Savings Sensitivity to Aerobrake Development Cost versus All Propulsive

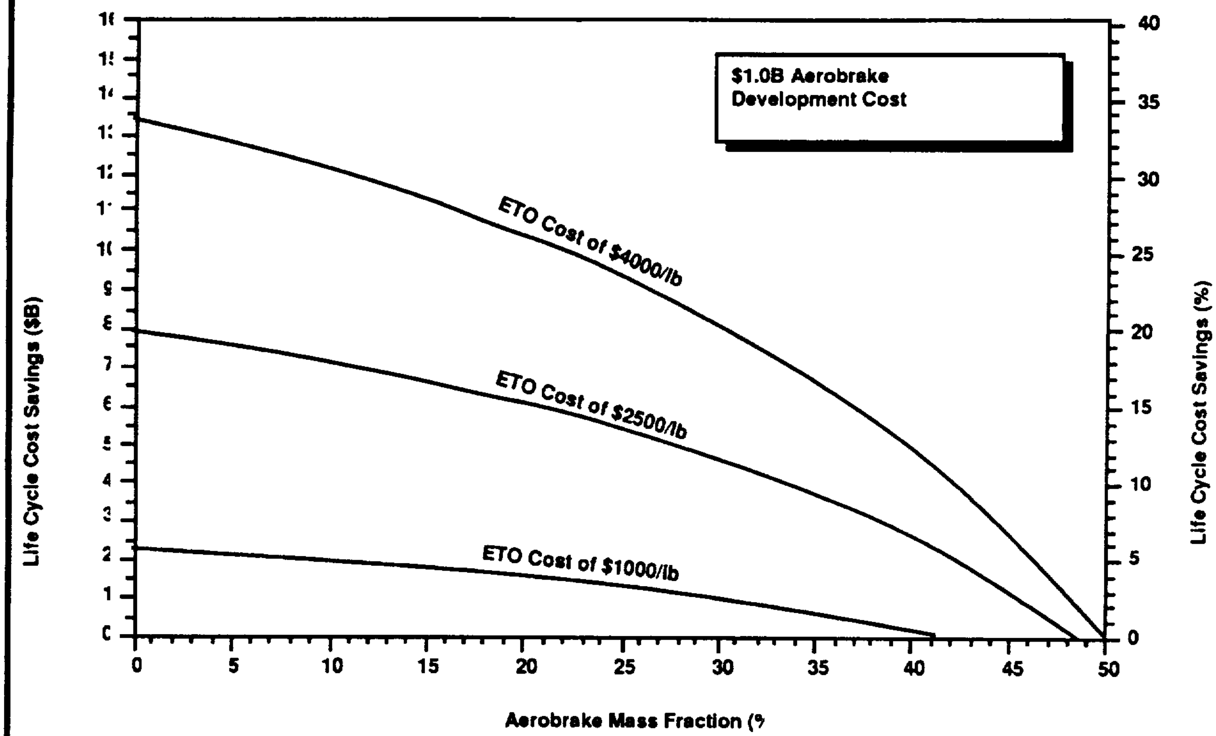


Figure 2.3.3.1-4 Cost Savings Sensitivity to ETO Cost for Aerobrake versus All Propulsive

Table 2.3.3.1-1 Groundrules and Assumptions

- Return to LEO From Lunar Mission
- Rigid AB, 5 Reuses
- Concept Single Proplusion Stage
  - Single Propulsion Module
  - Single Crew Compartment
  - AB Stays in LLO for Aeroassist Version
  - TEI/LEO Propellant Tanks Stay in LLO for All Propulsive Version
- ASE Engines; Isp = 476 sec.
- Piloted Vehicle Missions Only, 21 Flights
- 14.6 t Cargo in Addition to Crew
- ΔV from Aeroassist = 3150 M/Sec (10,332 ft/sec)
- AB Recurring Cost = $12M
- AB Development Cost = Variable
- ETO Cost ($/lb) = Variable
- AB Weight Fraction = Variable
- AB Weight Fraction Definition:

$$\frac{\text{AB Str/TPS Mass}}{\text{Total Entry Mass}}$$

Figure 2.3.3.1-2 illustrates the trends in numbers of missions required to amortize the development cost of the aerobrake as a function of the aerobrake mass fraction and the ETO delivery costs. The $0.5 B development cost was used arbitrarily and similar trends were seen for other values for development cost. Figures 2.3.3.1-3 and 2.3.3.1-4 illustrate the sensitivity of the cost savings over a 21 mission lifetime to aerobrake development costs and ETO delivery costs. Note that only at very low ETO costs and/or extremely high development costs does the crossover point with propulsive occur at low mass fractions.

This analysis was based on a specific set of groundrules and assumptions (Table 2.3.3.1-1) that were derived from the STV systems requirements and the lunar mission analysis data developed as part of STV mission analysis activity.

The analysis was based on the recommended single propulsion stage vehicle configuration that is defined in detail in the Concept Definition Section 3.0. Figure 2.3.3.1-5 provides a dimensional overview of the system during the reentry maneuver. Table 2.3.3.1-2 defines the mass properties associated with the vehicle.

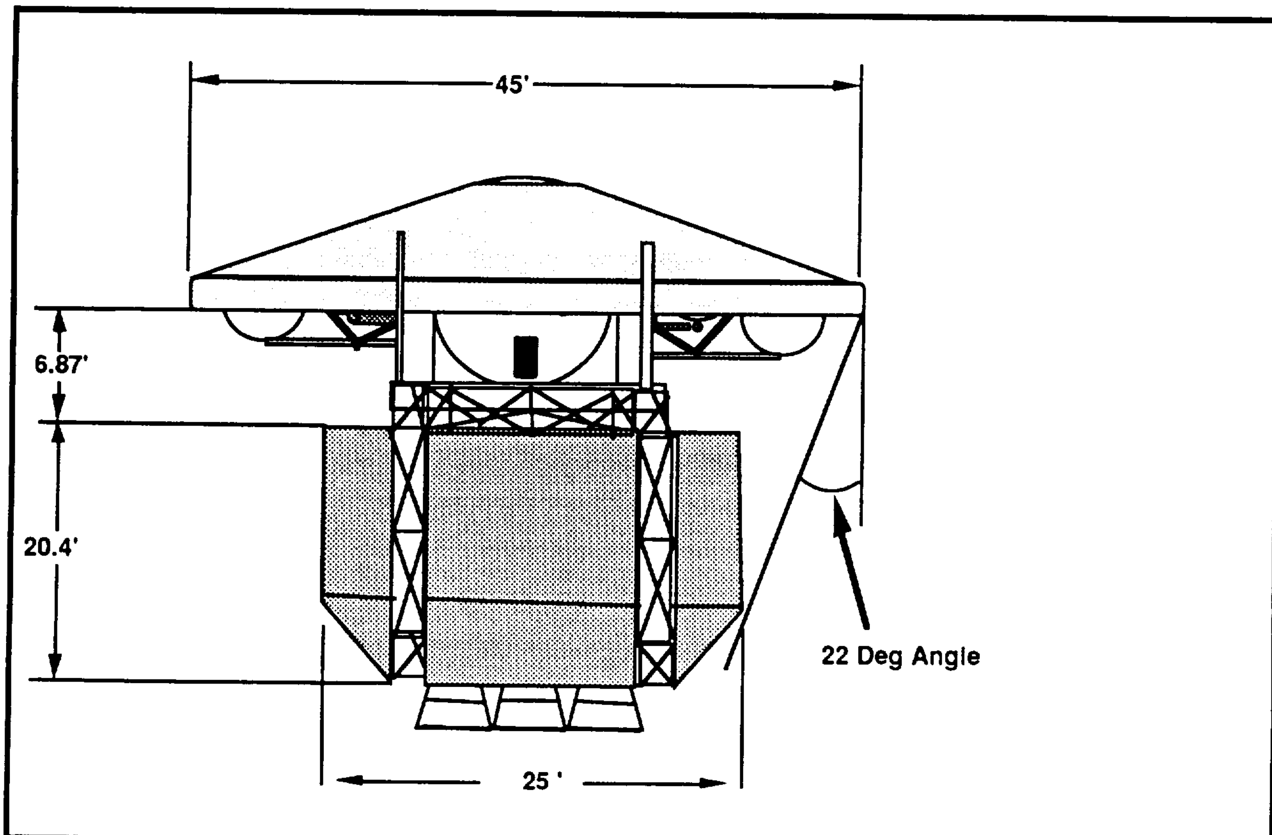


Figure 2.3.3.1-5 LTS Reentry Configuration

2.3.3.2 Space versus Ground Basing Analysis—The objective of the space versus ground basing analysis was to provide a means of course screening for the large configuration selection analysis, Section 2.3.3.3. This screen would allow the larger study to focus on one group of candidates, either space-based or ground-based configurations. To perform the analysis required, representative configurations for a space-based system and a ground-based system were defined as a result of information derived from the 1989 Skunk Works activities.

2.3.3.2.1 Groundrules and Assumptions—The following groundrules were observed in conducting this analysis:

- Propellant shall be cryogenic
- Earth return shall be aeroassisted (derived from results of the Aeroassist vs All-propulsive Return Study, Section 2.3.3.1),
- ASE engine shall be used on transfer vehicle (Isp - 476) and transfer/landing vehicle (Isp - 460),
- ETO transportation system cost shall be $2500/lb,
- LCC shall include design, development, test hardware and operations,
- System life shall be 30 years,
- Space basing shall utilize SSF, requiring $2.0 billion for modifications.

2.3.3.2.2 Cost and Operations Analysis—Cost and operations were the most important of the four primary analysis criteria under which the STV studies have been performed. Details of the configurations used to assess these criteria are shown in Figure 2.3.3.2-1. The space-based configuration is comprised of a multiple stage system with drop tanks for propellant storage. The ground-based system is comprised of an expendable transfer stage with a ballistic return lander and crew module. Details of both systems can be found in the configuration selection analysis section, 2.3.3.3, and the concept definition section, 3.0. Program cost defines the total cost to acquire and operated the system, including: Full Scale Development (FSD), verification, production, operations and support, and disposal. Operations analysis was primarily based on the complexity involved in performing the space flight phase of the lunar mission but also took into account some ground processing issues. Operational complexity is defined by the quantity as well as the complexity of the operational functions during the mission, with the emphasis placed on mission success and crew/cargo safety. The operational functions evaluated included rendezvous and docking both at Low Lunar Orbit (LLO) and Low Earth Orbit (LEO); engine burns at Trans-Lunar Injection (TLI), LLO, lunar landing, ascent, and Trans-Earth Injection (TEI); system element

Table 2.3.3.1-2 LTS Mass Properties

Propellants

Event	Aerobraked	All Propulsive
TLI	112.9	152.5
LOI	22.1	28.8
Descent/Ascent	31.7	31.7
TEI & EOI	7.3	30.7
Total	174.0	243.7

Aerobraked Vehicle

Element	Mass (t)	
Core	7.14	
TLI Tanks	5.72	
LOI Tanks	3.39	
Crew Module	7.79	
Crew & Consumables	0.66	
Aerobrake	3.50	
Structure & TPS		2.38
Tanks		0.26
RCS System		0.14
G. N. & C.		0.07
C .& D. H.		0.20
Electrical Power		0.41
Thermal Control		0.04
Total	28.2	

All Propulsive Vehicle

Element	Mass (t)	
Core	7.14	
TLI Tanks	7.72	
LOI Tanks	4.43	
Crew Module	7.79	
Crew & Consumables	0.66	
Return Stage (in LLO)	2.23	
Tanks & Structure		1.37
RCS System		0.14
G. N. & C.		0.07
C .& D. H.		0.20
Electrical Power		0.41
Thermal Control		0.04
Total	30.0	

separations including stages and drop tanks; crew, cargo and propellant transfers; and critical maneuvers including aerobrake preparation and operation and a ballistics return (Figure 2.3.3.2-2). Each of these functions was assigned either a Crit 1 or 2 rating, which provided a quantitative value to the criticality of the operation. A Crit 1 operation is defined as an operation which if not successfully completed results in loss of life or failure to deliver mission critical cargo. Crit 2 is defined as an operation which if not successfully completed allows the crew to return safely or leaves the cargo in a position where it can be salvaged.

2.3.3.2.3 Summary/Recommendation—The results of the cost evaluations are shown in Figure 2.3.3.2-3. This data shows that in three of the four cost categories the space-based systems represent a lower cost, including LCC. The only category in which the ground-based system rated better in cost was in DDT&E since the ground-based system uses fewer technology/advanced development items that require extensive development costs. The results of the operations evaluation show the opposite trend (Figure 2.3.3.2-4), with the ground-based system representing an approach with fewer critical failure modes during the conduct of the transfer missions. This can be attributed to fewer rendezvous and docking operations and the elimination of the aerobrake and the aeroassist maneuver. Further assessment of the operational complexity based on ground processing operations was conducted to cast a deciding vote in providing a recommendation from this analysis. This additional work indicated that the ground-based system greatly increased the processing requirements at KSC.

The recommended basing approach is to utilize a LEO transportation node and space-base the LTS. This provides an overall reduction in the system LCC of 9% and a similar approach to ground processing and launch at KSC. It should be noted that although this approach provides a lower cost, it does represent a system with more potential failure modes that must be accounted for in the final design.

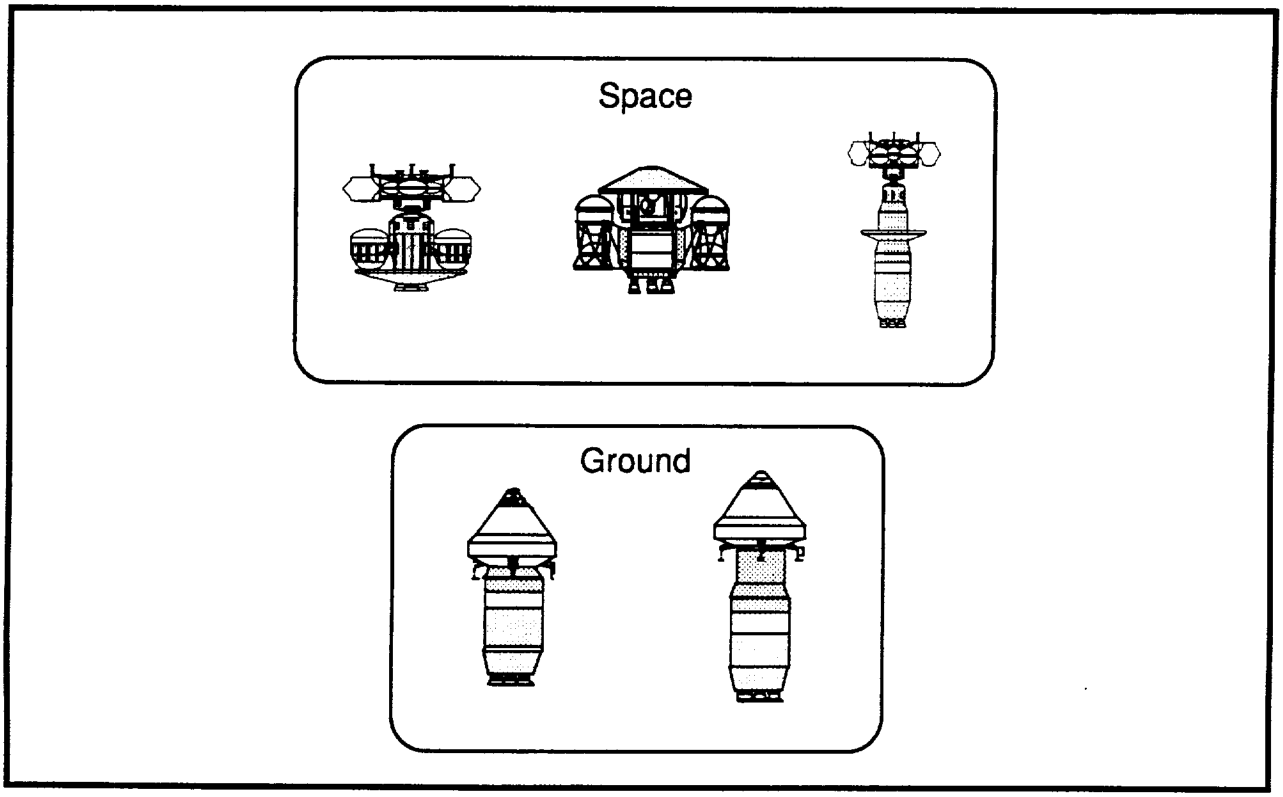


Figure 2.3.3.2-1 Basing Configuration Candidates

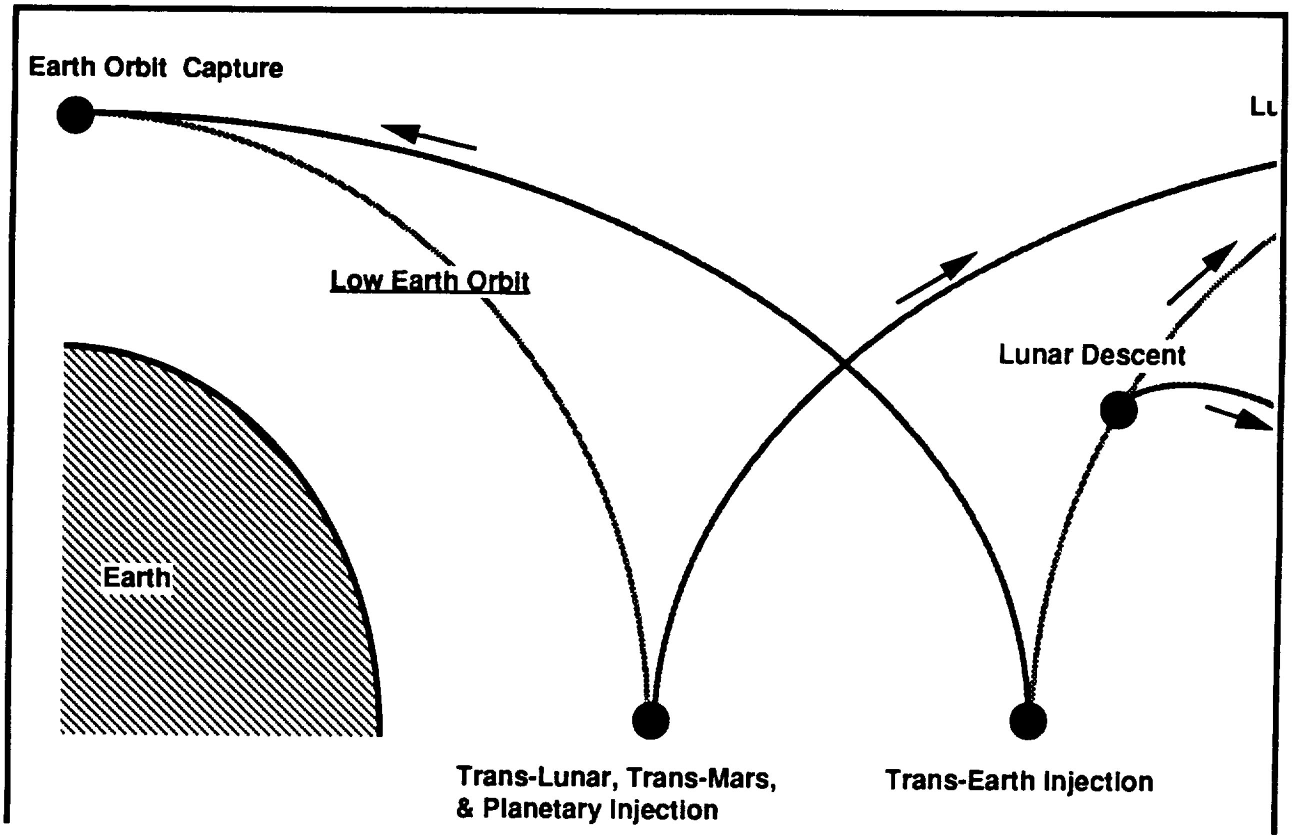


Figure 2.3.3.2-2 Lunar Mission Operational Functions

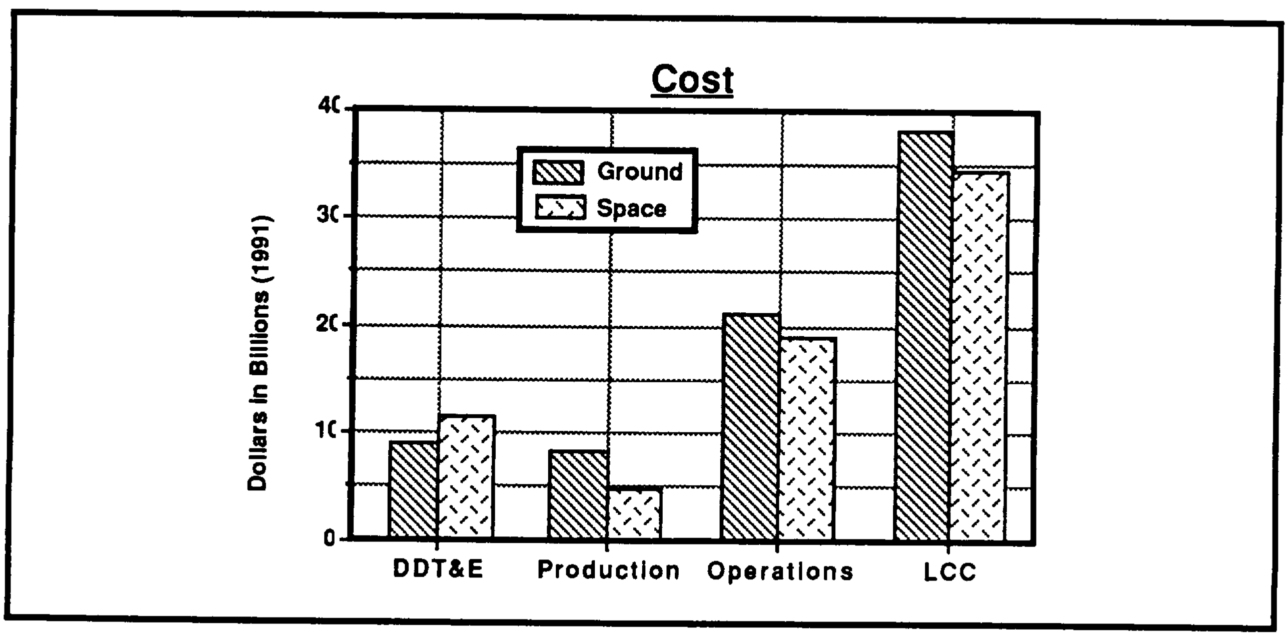


Figure 2.3.3.2-3 Basing Cost Analysis Results

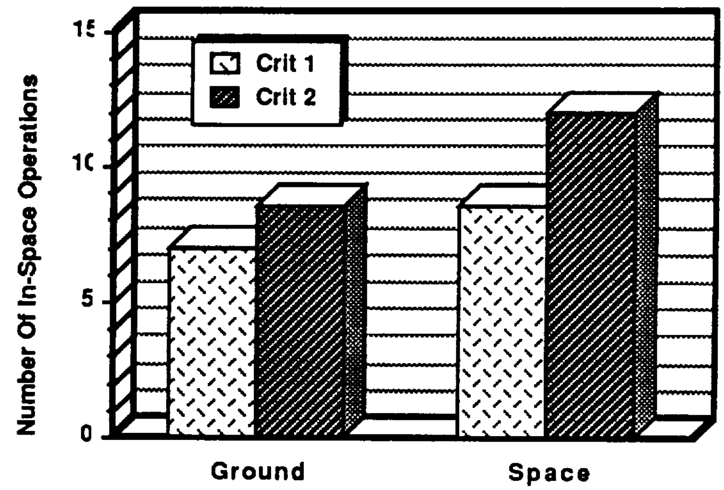


Figure 2.3.3.2-4 Basing Operations Analysis Results

2.3.3.3 STV Concept Selection Analysis—There are two basic STV concept selection philosophies. The first is to start with a ground-based initial STV, proceed to space-based reusable concepts, and continue to use the STV or family of STV vehicles for lunar missions and eventually Mars missions. A second philosophy starts with the most mission-driven STV concept — the lunar mission — and evolves backwards and forwards to satisfy the other missions. These two philosophies are illustrated in Figure 2.3.3.3-1. Since the lunar missions represent the most stringent drivers for vehicle definition, the concept selection philosophy of starting with the lunar STV family and evolving to the other design reference missions (DRMs) was utilized for this top level systems trade.

A concept selection process (Fig. 2.3.3.3-2) was established to systematically evaluate and downselect STV concepts into a single concept or family of concepts. The process began with the development of a concept selection methodology and was followed by a concept identification task. Once concepts were defined, simple configurations, operational scenarios, performance data and relative cost data were generated for each concept. Concepts were evaluated against top level selection criteria, performance, relative cost, and operational complexity.

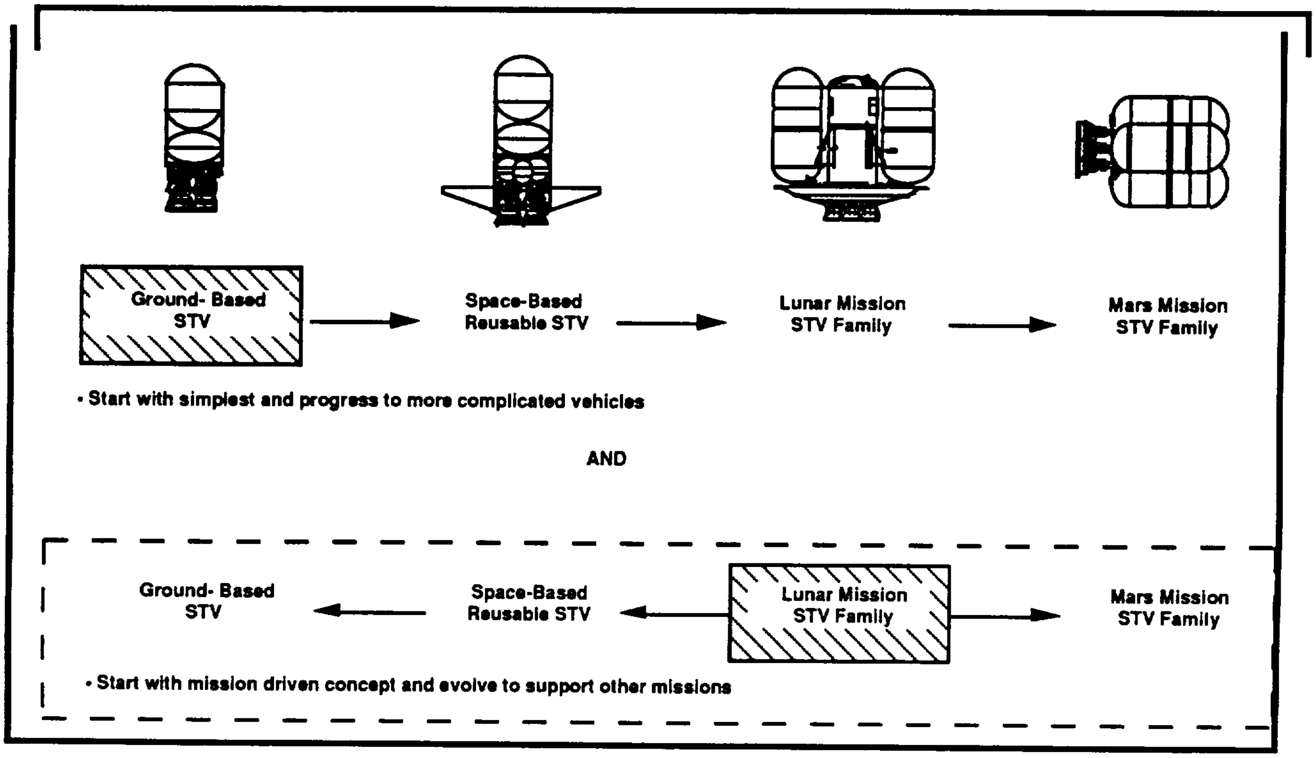


Figure 2.3.3.3-1 Concept Selection Philosophy

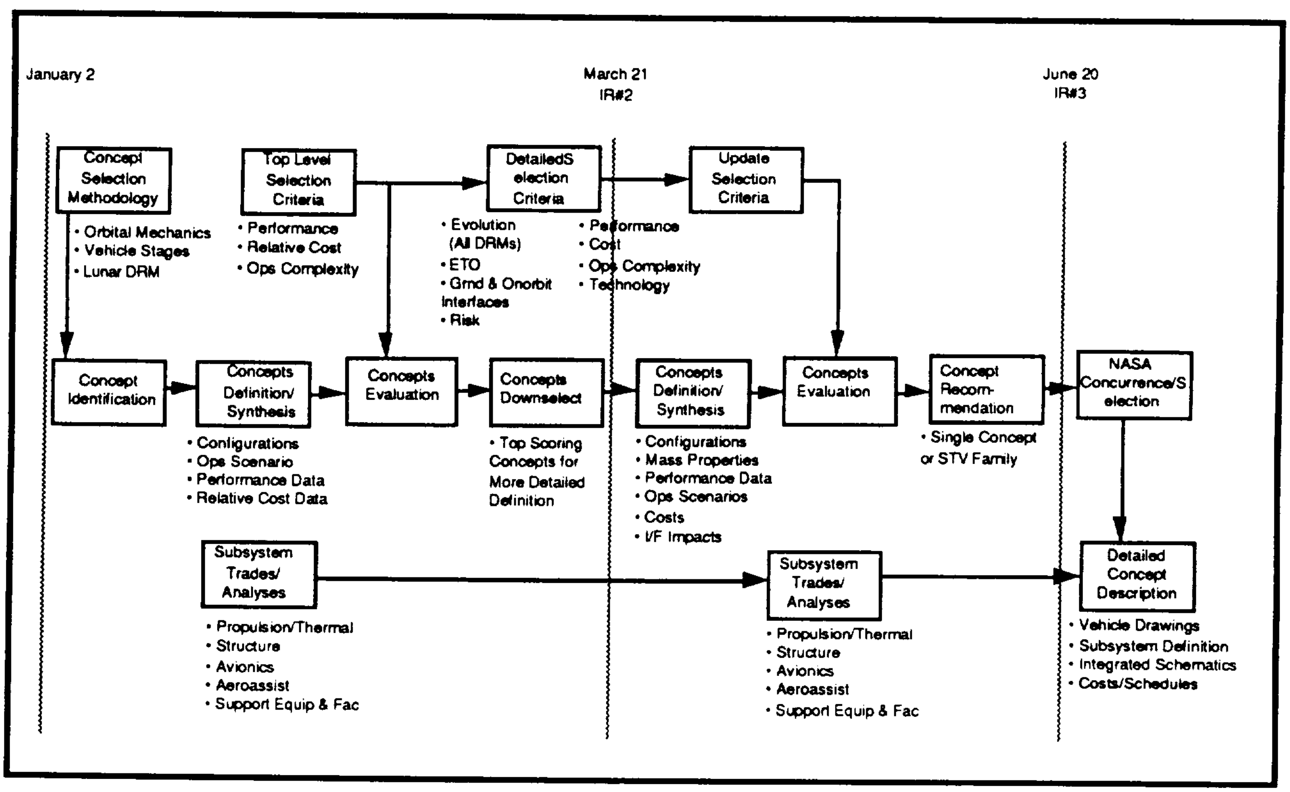


Figure 2.3.3.3-2 Concept Selection Process

Top scoring concepts for each selection criteria were recommended for additional evaluation during the second downselect process. Downselected concepts were further defined and evaluated to determine interface impacts, real costs, evolution to other missions, ETO transportation methods, etc.

After the first downselect, lunar architectures were developed and concepts were allocated to these architectures. More detailed data consisting of configurations, mass properties, performance results, flight operational scenarios, interface impacts and programmatic costs were generated for each concept. Cost, operations, adaptability to meet other DRMs, and risk were used as evaluation criteria to recommend criteria driven concepts for additional study during the final downselect.

The criteria-driven concepts were further studied to define a common family of vehicles and assess abort scenarios. Results from these final studies were evaluated, and a final STV family of vehicles was selected. Once NASA concurred with the final STV selection, results from subsystems trades were incorporated and detailed concept description of the selected concept and detailed programmatics were conducted.

2.3.3.3.1 First Downselect Process—

2.3.3.3.1.1 Methodology—A concept selection methodology as illustrated in Figure 2.3.3.3-3, was established to identify STV concepts and perform the first downselect process. The lunar missions were used as the driving missions. All possible orbital mechanics solutions to launch and/or return cargo and/or crew from the Earth to the Moon were identified. A matrix of possible launch and return options was developed and populated for reasonable mission scenarios. Several possible orbital mechanics solutions — libration points, cycler and HEO missions — were eliminated and removed from the matrix for reasons of excessive Δ-velocities, flight times, or excessive estimated cost. Remaining orbital mechanics solutions were input into a vehicle stage matrix of various transfer and landing vehicle options. Populating this matrix produced 10 cargo only (no return concepts) and 48 crew/cargo (reusable concepts). Crew/cargo concepts consisted of both single and dual crew cab options. Operations, relative cost, and performance data for each concept were then developed and evaluated. The top scoring concepts in each criteria were carried forward for more detailed evaluation during the second downselect phase.

2.3.3.3.1.2 Groundrules and Assumptions—The following top level groundrules and assumptions were used in the first downselect process:

- Lunar DRMs were used to define the initial concepts. Downselected concepts were evaluated for adaptability to meet all DRMs.
- Cargo only expendable concepts were rated separately from crew and cargo missions.
- Subsystem definition for all concepts were taken from the 90-Day Study baseline, i.e., rigid aerobrake, fuel cells, advanced space engines, etc.
- Initial vehicle delivery missions were evaluated for all concepts.
- Performance criteria scores were calculated by taking total propellant requirements and dividing by the cargo delivered — 33 tonnes for cargo and 14.6 tonnes for crew/cargo.
- No constraints were placed on the Earth-to-Orbit (ETO) transportation system or on using Space Station Freedom (SSF) as the transportation node.
- Only chemical LOX/LH_2 propulsion systems were considered.

The terms shown in Tables 2.3.3.3-1 and 2.3.3.3-2 were defined to provide insight to the data and rationale for the concept selection process. Definitions of stages, operations, elements, activities, etc. allow understanding of the terms used in the operational, performance, and relative cost evaluation.

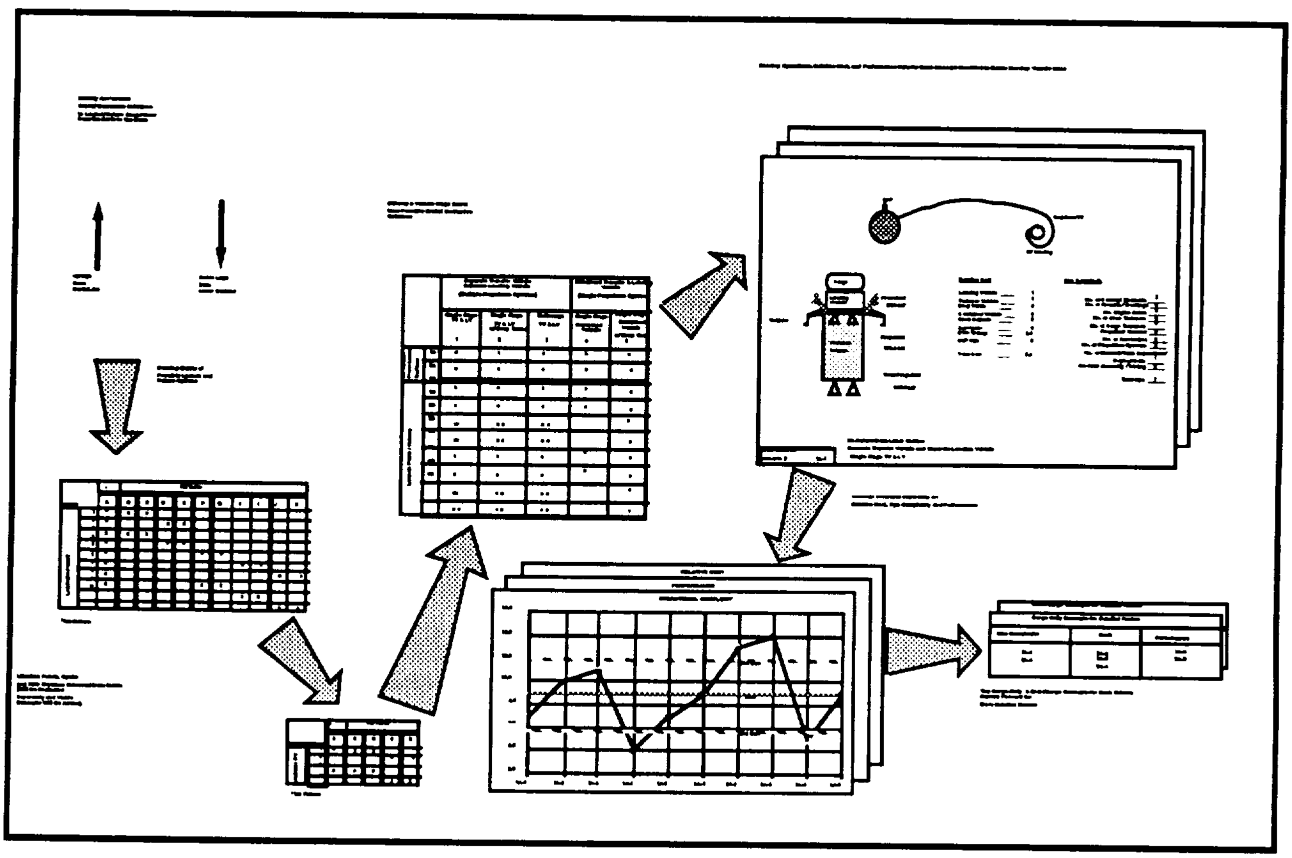


Figure 2.3.3.3-3 Concept Selection Methodology - 1st Downselect

2.3.3.3.1.3 Orbital Mechanics Solutions—The first step in the downselect process was to identify orbital mechanics solutions for delivering crew and/or cargo to the moon. Figure 2.3.3.3-4 is a pictorial overview of the node options available for lunar transfer and return. Nodes were defined as locations where two vehicles can meet to transfer people, cargo, and propellant. Low Earth Orbit (LEO) is often assumed to be the starting node, but for this trade direct ascents that pass directly through LEO on the way to the moon or an intermediate node were also considered. A Highly Elliptic [Earth] Orbit (HEO), an orbit with a perigee near Space Station Freedom (SSF) altitude and a period that is resonant with the sidereal rate of the moon, was evaluated as a node. L1, the libration point on a line between the Earth and moon, was also studied. L1 is the place where the net force of the two bodies' gravitational pulls exactly equal the centripetal acceleration associated with the moon's angular rate. L2 is a similar point considered, but located on the far

Table 2.3.3.3-1 Definition of Terms

Stage	**An element that consist of tanks and a propulsion system (may include an avionics system).**
ETO Launch Vehicle	**A stage that delivers STV and/or LEV elements to Earth suborbital or orbital altitude.**
Staging	**Separation of two stages (does not include drop tanks)**
Low Lunar Operations	**A sequence of events between two stages (rendezvous and docking) in low lunar orbit.**
Combined STV/LEV	**A vehicle that has a single propulsion system.**
Separate STV & LEV	**Individual vehicles that are separated and/or docked during the course of the mission each having its own propulsion system.**
LEO Operations	**A sequence of events performed in low earth orbit which may occur at SSF, at a separate platform or at a free flying platform.**
No Return	**No vehicle and/or equipment is returned from the lunar surface for that mission.**
Concept Element	**Stages, drop tanks, transfer and landing vehicles. Crew modules included only when required to separate from TV and dock with LV. Drop tanks count as one element. (Does not include cargo.)**

Table 2.3.3.3-2 Definition of Terms

Rendezvous/Docking	**An operation that places two elements in close proximity and accomplishes physical connection in flight.**
Engine Burns	**Firing of main engines on a stage. (Does not include RCS.)**
Crew Transfers	**Transfer of crew from one element to another by EVA or IVA. (Includes transfers to and from crew module in LEO. Does not include crew EVA on Lunar surface.)**
Cargo Transfer	**Mating of cargo to TV or LV. (Does not include deployment of cargo on Lunar surface)**
Propellant Transfer	**Transfer of propellant from one element to another.**
Aerobrake	**Breaking of Return Vehicle either by aerobraking or ballistic heat shield. (All return flights have one aerobrake.)**
Propulsion System	**One or more engines that provide primary propulsion to a stage.**
Separations/ Deployment	**Separation of elements such as dropping empty propellant tanks, separating LV from TV and separating crew modules.**
On Orbit Assembly/ Mating	**Mating of elements in LEO. (Does not include cargo).**
Expendable Element	**Any element that does not return to its launch site.**

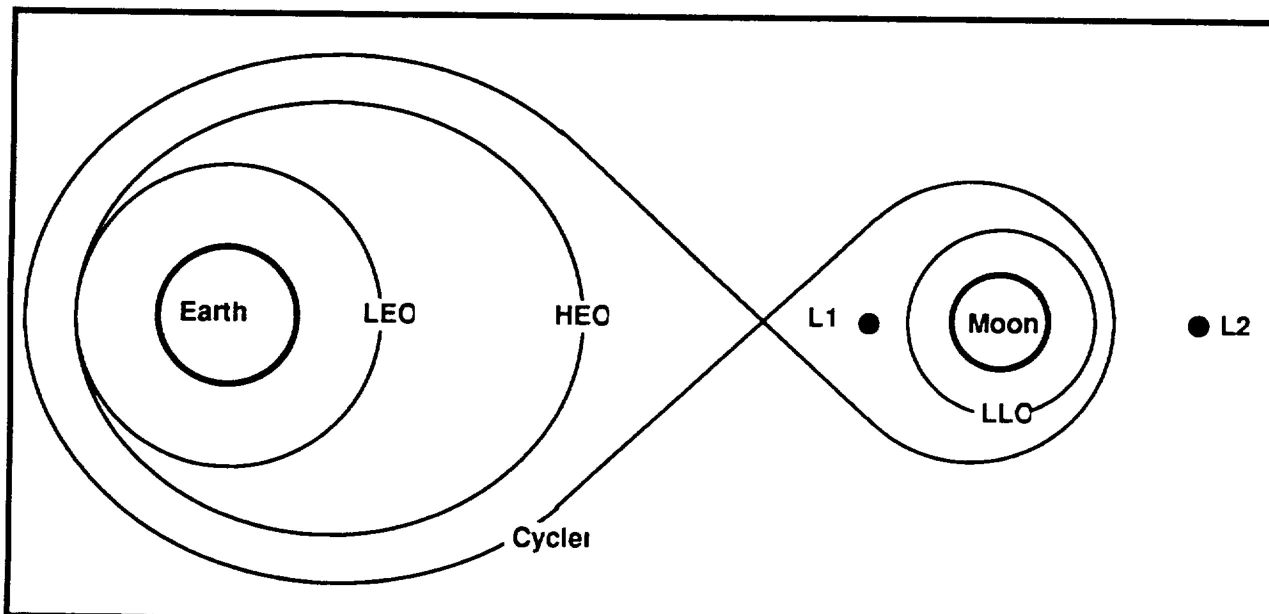


Figure 2.3.3.3-4 Lunar Mission Orbital Mechanics Options

side of the moon, still on the Earth-moon line. At L2, the combined pull of Earth and moon are balanced by the greater centripetal acceleration of being farther from the center of rotation of the

Earth-Moon system. A cycler, which is a continually moving node that is placed in a resonate, free-return trajectory between the Earth and moon, was also defined. Crew transfer from SSF to the cycler occurs when the cycler swings-by LEO in a small "taxi" vehicle. Once on the cycler, the crew transfers to the cycler habitation modules and ride to the Moon in more spacious accommodations. At the moon, t he crew transfers to the lander vehicle docked to the cycler and departs for the lunar surface. The reverse process is followed for getting back to Earth. The final node considered was Low Lunar Orbit (LLO), typically a 300 kilometer circular orbit with an inclination of less than 30 degrees.

Using the node options for lunar transfer and return as described above, all possible orbital mechanics solutions to launch and/or return cargo and/or crew from the Earth to the Moon were developed and are listed below:

Launch - Up Leg from Earth or Low Earth Orbit (LEO) to Lunar Surface

1	Earth to Lunar Surface
2	Earth to Low Lunar Orbit (LLO) to Lunar Surface
3	LEO to Lunar Surface
4	LEO to LLO to Lunar Surface
5	Earth to Libration Point to Lunar Surface
6	Earth to Highly Elliptic [Earth] Orbit (HEO) to Lunar Surface
7	Earth to Cycler to Lunar Surface
8	LEO to Libration Point to Lunar Surface
9	LEO to HEO to Lunar Surface
10	LEO to Cycler to Lunar Surface

Return - Down Leg from Lunar Surface

A	No Return
B	Direct Return from Lunar Surface to Earth
C	Direct Return from Lunar Surface to LEO
D	From Lunar Surface to LLO to Earth
E	From Lunar Surface to LLO to LEO
F	From Lunar Surface to Libration Point to Earth
G	From Lunar Surface to Libration Point to LEO
H	From Lunar Surface to HEO to Earth
I	From Lunar Surface to HEO to LEO
J	From Lunar Surface to Cycler to Earth

K From Lunar Surface to Cycler to LEO

Launch Pt - Up Leg From Earth / Leo

1 Earth To Lunar Surface
2 Earth To LLO Ops To Lunar Surface
3 LEO To Lunar Surface
4 LEO To LLO Ops To Lunar Surface
5 Earth To Libration Pt. To Lunar Surface
6 Earth To HEO To Lunar Surface
7 Earth To Cycler To Lunar Surface
8 LEO To Libration Pt. To Lunar Surface
9 LEO To HEO To Lunar Surface
10 LEO To Cycler To Lunar Surface

Return - Down Leg From Lunar Surface

A: No Return
B: Direct Return From Lunar Surface To Earth
C: Direct Return From Lunar Surface To LEO
D: From Lunar Surface To LLO Ops To Earth
E From Lunar Surface To LLO Ops To LEO
F From Lunar Surface To Libration Point To Earth
G: From Lunar Surface To Libration Point To LEO
H: From Lunar Surface To HEO To Earth
I From Lunar Surface To HEO To LEO
J: From Lunar Surface To Cycler To Earth
K: From Lunar Surface To Cycler To LEO

	*	RETURN									
LAUNCH POINT	A	B	C	D	E	F	G	H	I	J	K
1	X	X	X								
2	X			X	X						
3	X	X	X								
4	X			X	X						
5	X					X	X				
6	X							X	X		
7	X									X	X
8	X					X	X				
9								X	X		
10										X	X

* No Return

Figure 2.3.3.3-5 Orbital Solution Matrix

These orbital mechanics launch/return options were used to populate and develop a matrix of reasonable orbital solutions for the lunar mission as shown in Figure 2.3.3.3-5 Return options were assumed to follow the same path as launch options, i.e., an option that went through libration points on the up leg must return through the libration points on the return leg.

In order to reduce the number of orbit mechanics approaches, the Δ-velocities required to complete either a one-way or round-trip mission to the Moon were calculated. All node options were considered except the cycler option which was eliminated on assumed cost grounds and operational complexities associated with lunar-to-Earth return and abort scenarios. Five transfer options are shown in Figure 2.3.3.3-6: one is direct, two go through LLO with varying transfer speeds, and two use the first and second Earth-Moon libration points. Aerobraking was assumed for all cases having Earth returns. The lowest Δ-velocity option is the direct transfer from LEO to the lunar surface and return. The next most efficient transfer is to go through LLO on either a 3-day or minimum energy/5 day trajectory. The round trip Δ-velocity is 360 m/s higher when the trip time is dropped from 5 to 3 days and the 5-day trajectory is 294 m/s higher than a 3-day direct transfer.

The libration point nodes are much less efficient because the Δ-velocities required to get into and out-of the libration point. L1 takes 860 m/s to arrive or depart. L2 is easier because of the advantage of lunar swing-by but still takes 300 m/s to arrive or depart and takes another 80 m/s to get to it from LEO. Because L1 and L2 required more Δ-velocity, they were eliminated as viable options.

The HEO node scenario offers some advantages over using LLO — namely reduced Δ-velocity budget for the Lunar Transfer Vehicle (LTV). Using LLO, the LTV sees about 5800 m/s; however, using HEO (close to escape energy), the LTV sees only 3800 m/s. Since perigee of HEO is around 500 nmi, very little delta-V is required to set up an aeropass to return the vehicle back to LEO. The LTV becomes more of a booster stage than a lunar transfer stage. The increased Δ-velocity budget for the Lunar Excursion Vehicle (LEV) goes from about 4000 m/s to approximately 6000 m/s. Hence a more even split of Δ-velocities is achieved and the LEV is more flexible on landing sites. The down side of the HEO node scenario is the reduced return opportunities from the Moon. The HEO orbit is set up as a submultiple of the lunar orbit period. A submultiple of 4 means a spacecraft in HEO will orbit four times in 27.3 days, the sidereal

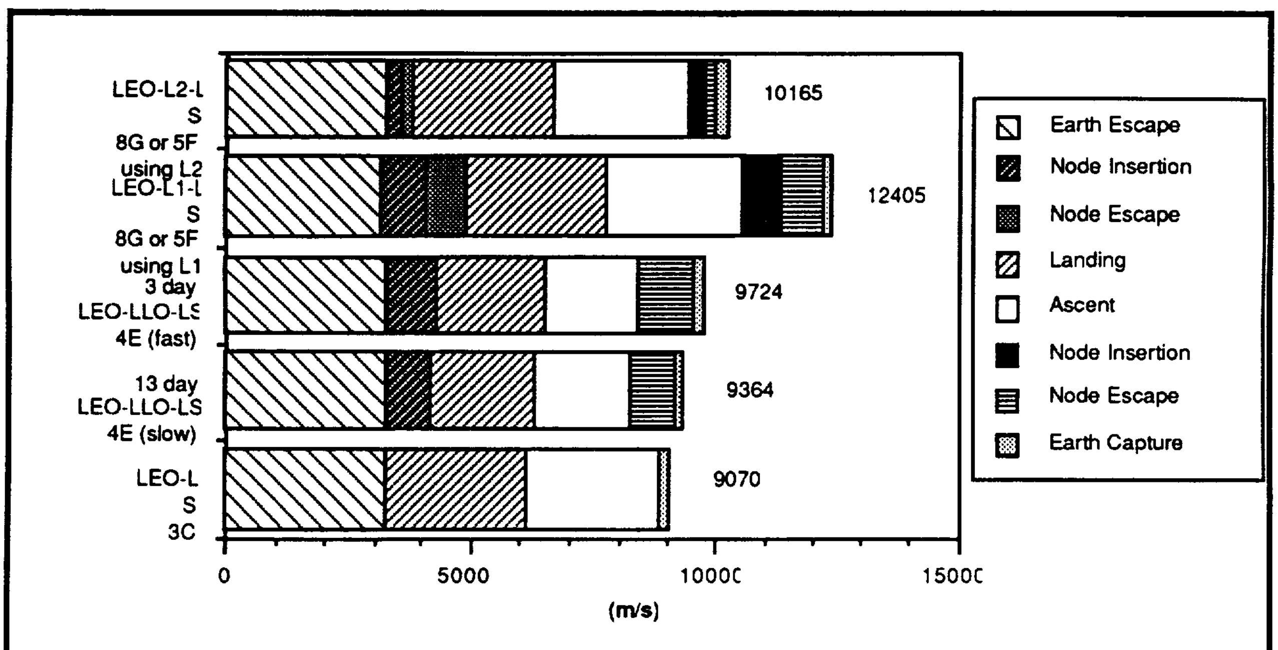


Figure 2.3.3.3-6 Transfer Option Δ-Velocities

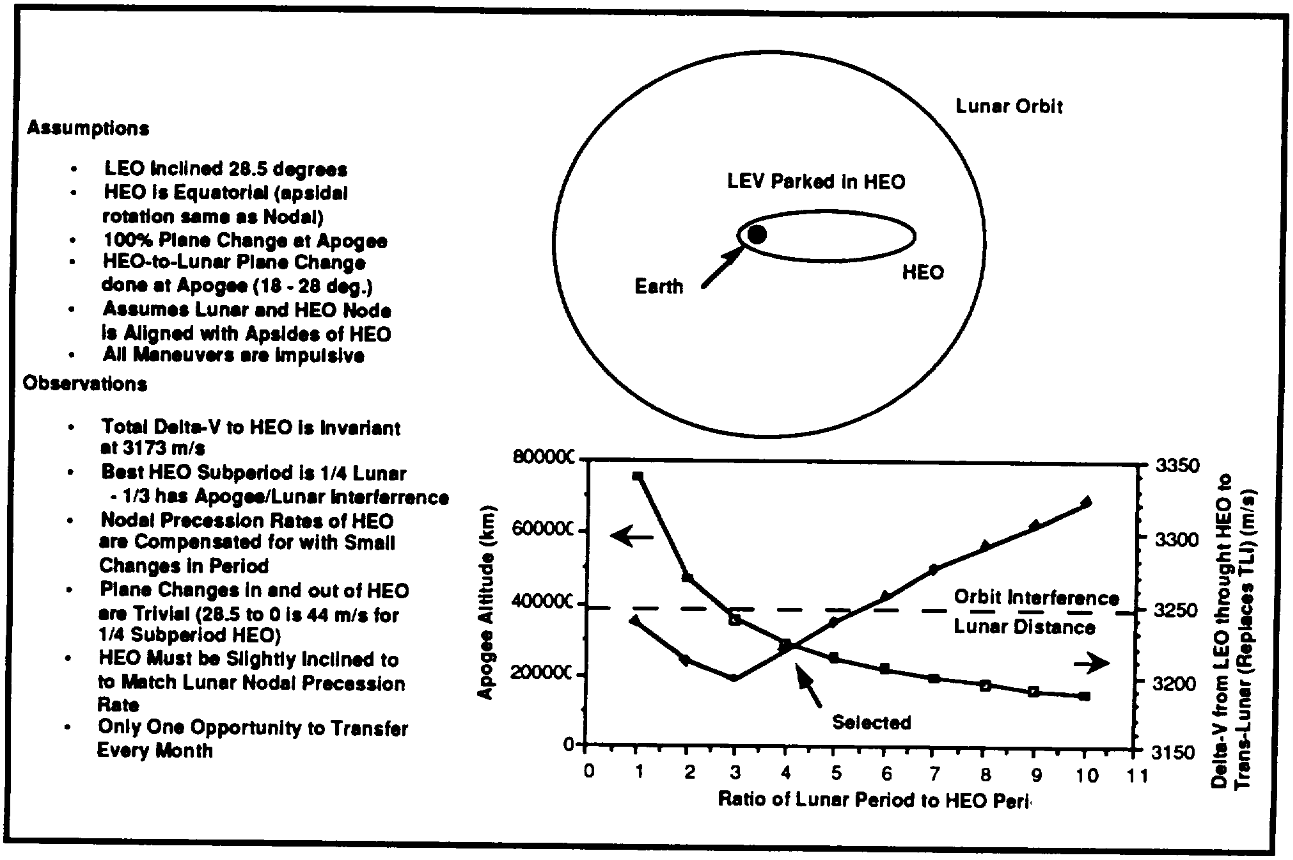


Figure 2.3.3.3-7 HEO Docking & Payload Transfer

period of the Moon. As illustrated in Figure 2.3.3.3-7, the best submultiple is 3 in terms of Δ-velocity, but this interferes with the moon and is not stable. The next best submultiple is 4 with an apogee of 280,000 km that is not greatly perturbed by the moon. Although the LEV can rendezvous with HEO at any time, only once each month will the Δ-velocities be minimum. This is true no matter what submultiple is selected. Using a submultiple of 4 gives a total Δ-velocity of 3220 m/s to go from LEO to trans-lunar. Comparing this Δ-velocity against the 90-day reference approach of 3100 m/s shows little penalty for going through HEO. The other consideration of HEO is the three-dimensional aspects of the orbits. LEO is inclined at 28.5 degrees and the moon ranges between ± 18 to 28 degrees. Determining the inclination of HEO and what the nodal and apsidal precession rates do to the Δ-velocities to get in and out of HEO are complicated except for equatorial cases. For an equatorial HEO, the apsidal rotation combines with the line of nodes regression so that all one has is a rotation of the line of apsides in inertial longitude relative to the Earth's equator. For an equatorial HEO, the plane changes from LEO and the Moon are very large (about 28 degrees for each), but is insignificant because at HEO apogee the spacecraft is only traveling at 245 m/s. The apsidal rotation rate of an equatorial HEO will only affect the repeat period between it and either LEO or the Moon. For LEO, the repeat rate is once every 25.5 days and for the Moon, the repeat rate is every 27.3 days. In summary from an opportunity point of view, HEO has distinct disadvantages over direct transfers and was therefore eliminated from further evaluation.

A downscaled orbital mechanics matrix of reasonable orbital mechanics solutions for the lunar mission (Fig. 2.3.3.3-8) was developed after the libration point, HEO, and cycler options were removed. The remaining solutions use only Earth and the LEO node as starting points for mission activities.

The remaining orbital mechanics solutions were combined with possible vehicle stage solutions to create a vehicle stage matrix (Fig. 2.3.3.3-9). Separate single stage landing and transfer vehicles with and without drop tanks, multistage vehicles, and single stage combined vehicles with and without drop tanks were considered. Options for single or dual crew cabs were also included for several crew/cargo reusable options. Using this matrix approach, 10 cargo only options were identified and 48 crew/cargo options were identified.

2.3.3.3.1.4 Operational, Performance & Relative Cost Data—Preliminary operational scenarios and vehicle configurations for each possible concept solution from the matrix were developed. Performance analyses were run to determine vehicle propellant quantities required to deliver 33 tonnes of cargo for the no return concepts and 14.6 tonnes of crew/cargo for the manned return

Launch Pt. - Up Leg From Earth / Leo

1 Earth To Lunar Surface
2 Earth To LLO Ops To Lunar Surface
3 LEO To Lunar Surface
4 LEO To LLO Ops To Lunar Surface

Return - Down Leg From Lunar Surface

A: No Return
B: Direct Return From Lunar Surface To Earth
C: Direct Return From Lunar Surface To LEO
D: From Lunar Surface To LLO Ops To Earth
D: From Lunar Surface To LLO Ops To LEO

	*	RETURN			
LAUNCH PT	A	B	C	D	E
1	X	X	X		
2				X	X
3	X	X	X		
4				X	X

* No Return

HEO, Cycler, and Libration Point Solutions Deleted from Matrix

Figure 2.3.3.3-8 Downscaled Orbital Mechanics Matrix

LAUNCH PT	* A	RETURN B	C	D	E
1	X	X	X		
2				X	X
3	X	X	X		
4				X	X

* No Return

Launch Pt. - Up Leg From Earth / Leo

1: Earth To Lunar Surface
2: Earth To LLO Ops.. To Lunar Surface
3: LEO To Lunar Surface
4: LEO To LLO Ops.. To Lunar Surface

Return - Down Leg From Lunar Surface

A: No Return
B: Direct Return From Lunar Surface To Earth
C: Direct Return From Lunar Surface To LEO
D: From Lunar Surface To LLO Ops.. To Earth
E: From Lunar Surface To LLO Ops.. To LEO

		Separate Transfer Vehicle Separate Landing Vehicle (Multiple Propulsion Systems)			Combined Transfer & Landing Vehicle (Single Propulsion System)	
		Single Stage TV & LV 1	Single Stage TV & LV w/ Drop Tanks 2	Multistage TV & LV 3	Single Stage Combined Vehicle 4	Single Stage Combined Vehicle w/Drop Tks.
No Return From Lunar Surface	1A	x	x	x	x	x
	3A	x	x	x	x	x
Launch Point / Return	1B	x	x	x	x	x
	1C	x	x	x	x	x
	2D	x x	x x	x x	N/A	x
	2E	x x	x x	x x	N/A	x
	3B	x	x	x	x	x
	3C	x	x	x	x	x
	4D	x x	x x	x x	N/A	x
	4E	x x	x x	x x	N/A	x

Note: Two X's Indicate Single and Dual Crew Cab Concepts

Figure 2.3.3.3-9 Vehicle Stage Matrix

concepts. Each concept was also evaluated for operational complexity by determining the number of elements, operations/maneuvers, transfers, matings, separations, etc. Relative cost data was generated for each concept by determining the number of elements, ETO transportation requirements based on using a 150 klb launch vehicle, and SSF operations. An evaluation sheet as shown in Figure 2.3.3.3-10 was developed for each concept. Data from these sheets were used to identify trends and to downselect concepts for additional study. Sheets for all concepts considered in the study are included as Appendix 1.

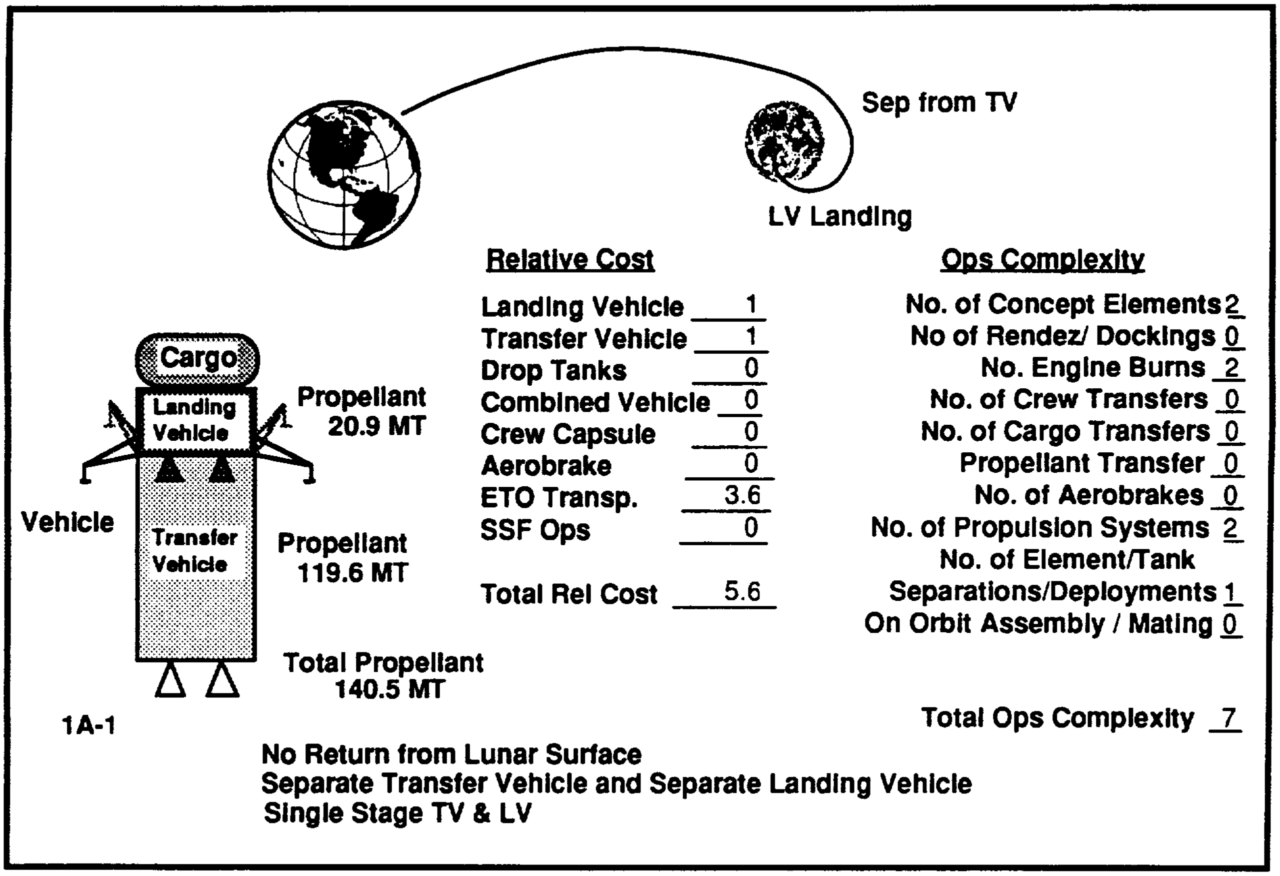


Figure 2.3.3.3-10 Typical Evaluation Sheet

2.3.3.3.1.5 Concept Trends—Trends in operational complexity for the cargo only expendable concepts are illustrated by the bar charts shown in Figure 2.3.3.3-11. The complexity increases as the vehicle configurations go from single stage combined vehicles to single stage separate vehicles to multistage vehicles. Drop tanks add the complexity of extra elements, propellant transfer, and tank separations. Operational complexity also increases for those missions that utilize LEO operations.

Trends in operational complexity, propellant quantity and relative costs for the cargo only expendable concepts are illustrated by the bar charts shown in Figure 2.3.3.3-1. The operational complexity increases as the vehicle configurations go from single stage combined vehicles to single stage separate vehicles to multistage vehicles. The opposite trend is noted for propellant quantity – multistage vehicles use less propellant than separate vehicles or single combined vehicles. Relative cost exhibits a trend similar to operational complexity.

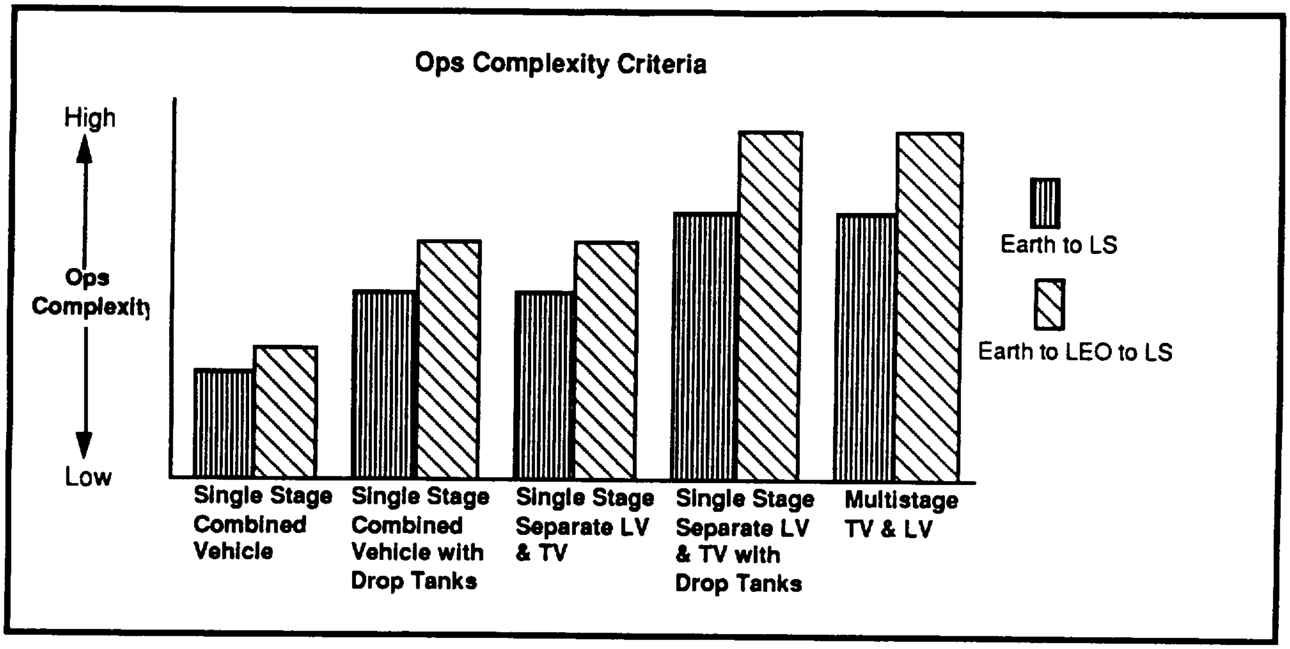


Figure 2.3.3.3-11 Cargo Only - Ops Complexity Trends

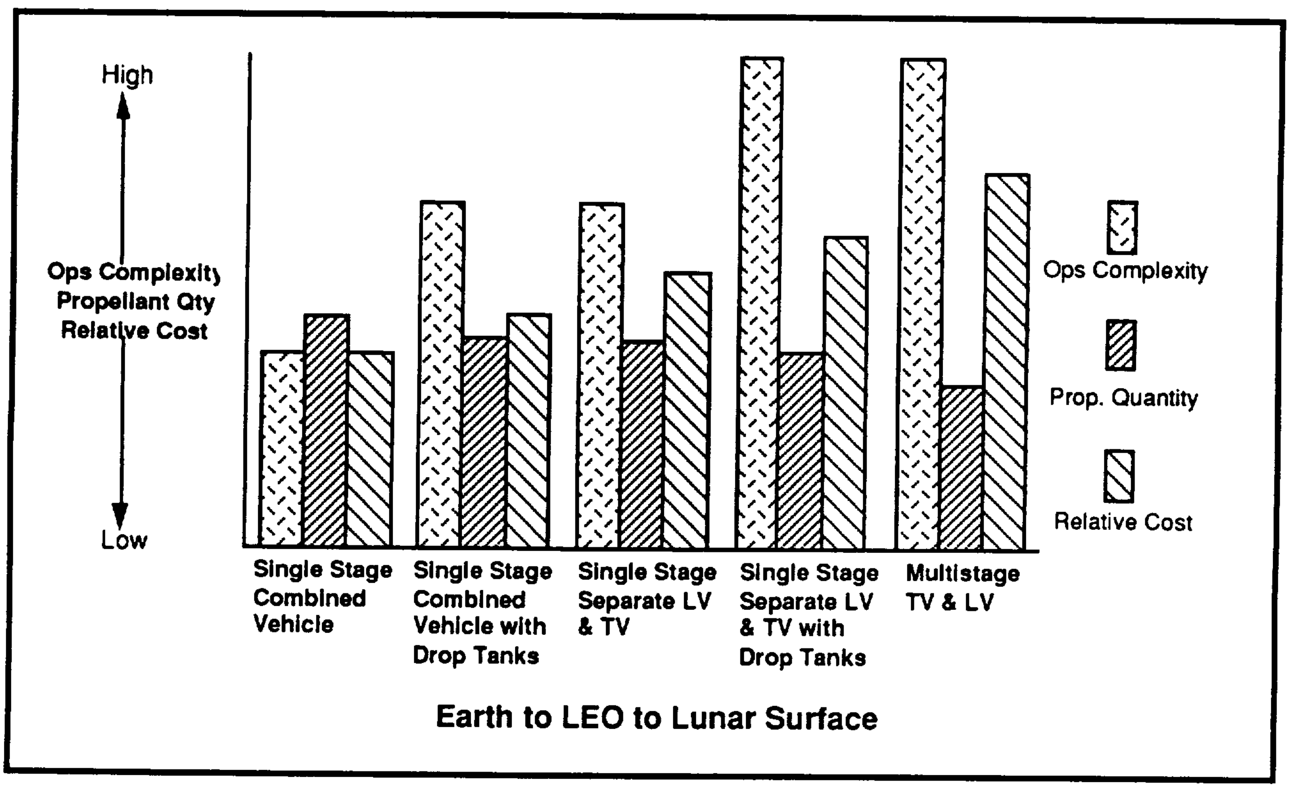
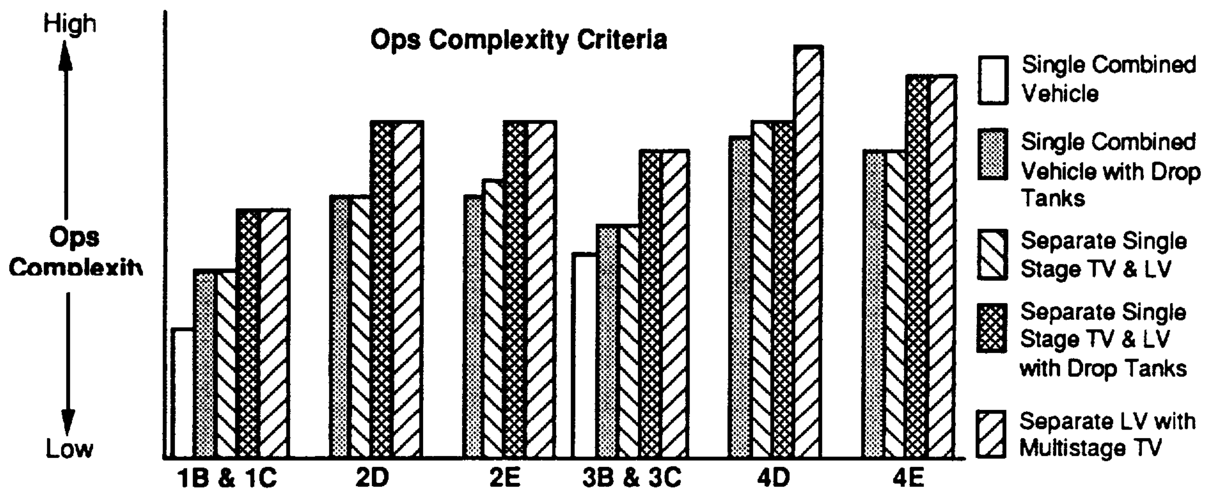


Figure 2.3.3.3-12 Cargo Only - Operations, Propellant, and Cost Trends

Trends in operational complexity for the crew/cargo return concepts are illustrated in Figure 2.3.3.3-13. As in the cargo only, the complexity increases as the vehicle configurations go from single stage combined vehicles to single stage separate vehicles to multistage vehicles. Drop tanks add the complexity of extra elements, propellant transfer, and tank separations. Operational complexity also increases from simple missions that go directly from Earth to the lunar surface to missions that use Space Station Freedom in LEO to missions that use both LEO and LLO operations.



Launch PT.-Up Leg From Earth/LEO	Return - Down Leg From Lunar Surface
1. Earth to Lunar Surface	A. No Return
2. Earth To LLO Ops To Lunar Surface	B. Direct Return From Surface To Earth
3. LEO To Lunar Surface	C. Direct Return From Lunar Surface To LEO
4. LEO To LLO Ops To Lunar Surface	D. From Lunar Surface To LLO Ops To Earth
	E. From Lunar Surface To LLO Ops To LEO

Figure 2.3.3.3-13 Crew/Cargo - Operations Complexity Trends

Trends in propellant quantity for single versus two crew cabs for the crew/cargo return concepts is shown in Figure 2.3.3.3-14. For all concepts, the single crew cab configurations require more propellant because a heavier crew cab is being taken to the lunar surface and must be returned from the lunar surface.

Figures 2.3.3.3-15 and 2.3.3.3-16 show the trends in operational complexity, propellant quantity and relative costs for the crew/cargo return concepts. The operational complexity increases as

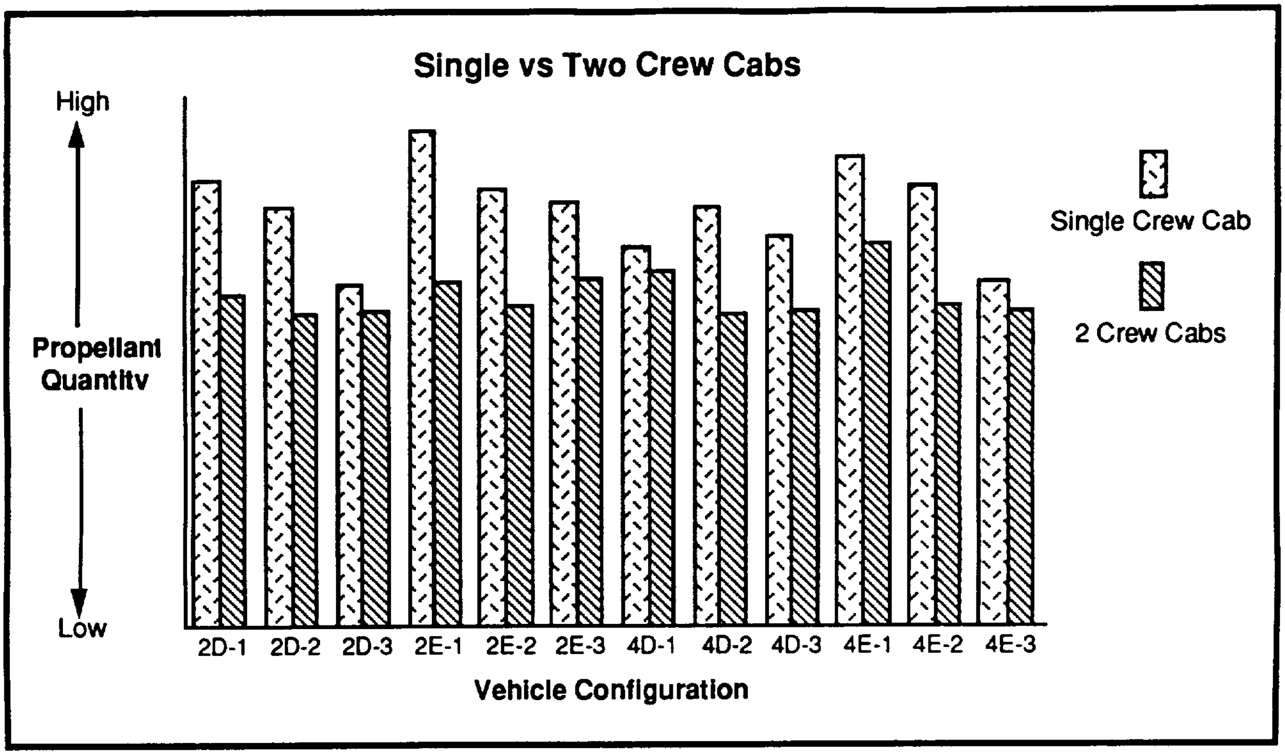


Figure 2.3.3.3-14 Crew/Cargo Cab Trends

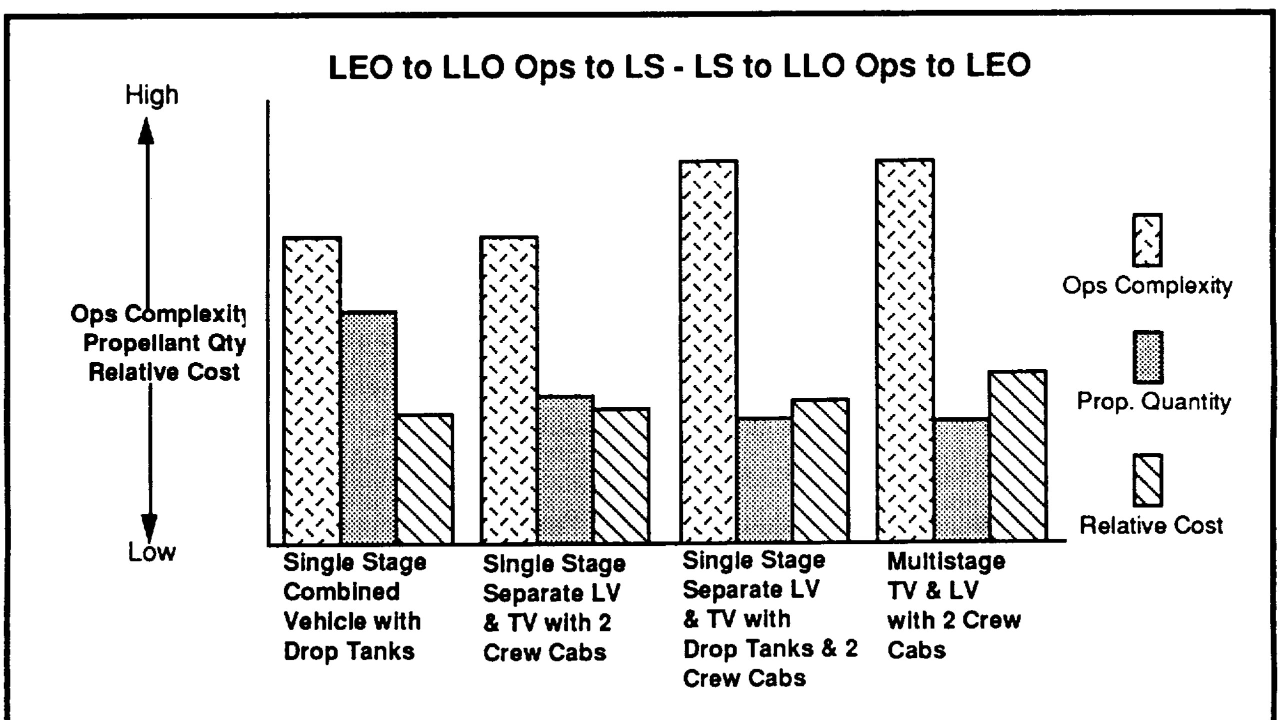


Figure 2.3.3.3-15 Crew/Cargo Operations, Propellant, and Cost Trends

the vehicle configurations go from single stage combined vehicles with drop tanks to single stage separate vehicles to multistage vehicles. The opposite trend is noted for propellant quantity – multistage vehicles use less propellant than separate vehicles or single combined vehicles with drop tanks. Relative cost exhibits a similar trend as operational complexity. The operational complexity, propellant quantity and relative costs increase when the mission uses LEO operations regardless of vehicle stage configuration.

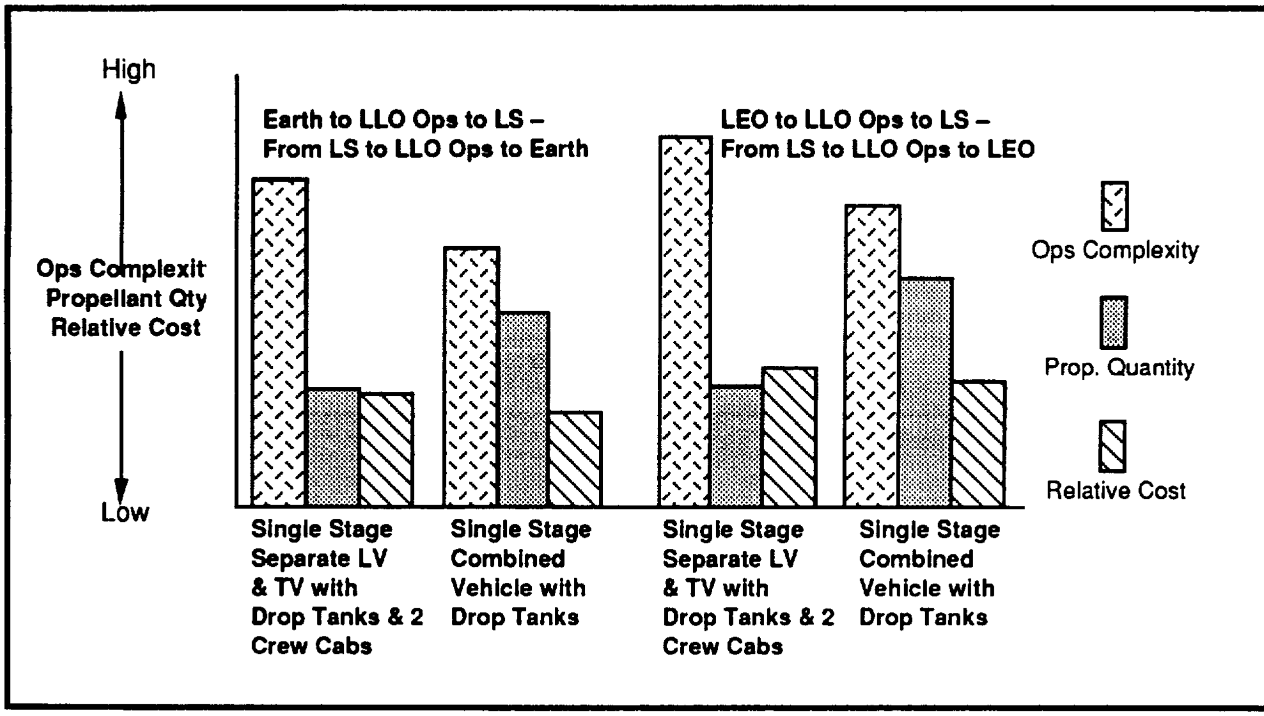


Figure 2.3.3.3-16 Crew/Cargo LEO Operations Trends

2.3.3.3.1.6 Summaries/Recommendations — Summaries of the concept criteria data, operational complexity, performance, and relative cost for the cargo only concepts are illustrated in Figures 2.3.3.3-17 through -19. The average criteria scores for these concepts along with the one sigma standard deviations are plotted on each chart. Those concepts that fall below the standard deviation were recommended for additional review and study. For operational complexity, the single stage combined vehicles scored the best with concepts 1A-4 and 3A-4 being the best with concepts 1A-3 and 3A-3 being downselected for additional study. For the final criteria of relative cost, single combined vehicles with and without drop tanks scored the best with concepts 1A-4 and 1A-5 being downselected for more study.

The recommended concepts for the cargo only scenario are shown in Figure 2.3.3.3-20. Two concepts use multistage transfer vehicles, two use a single combined vehicle, and one uses a single

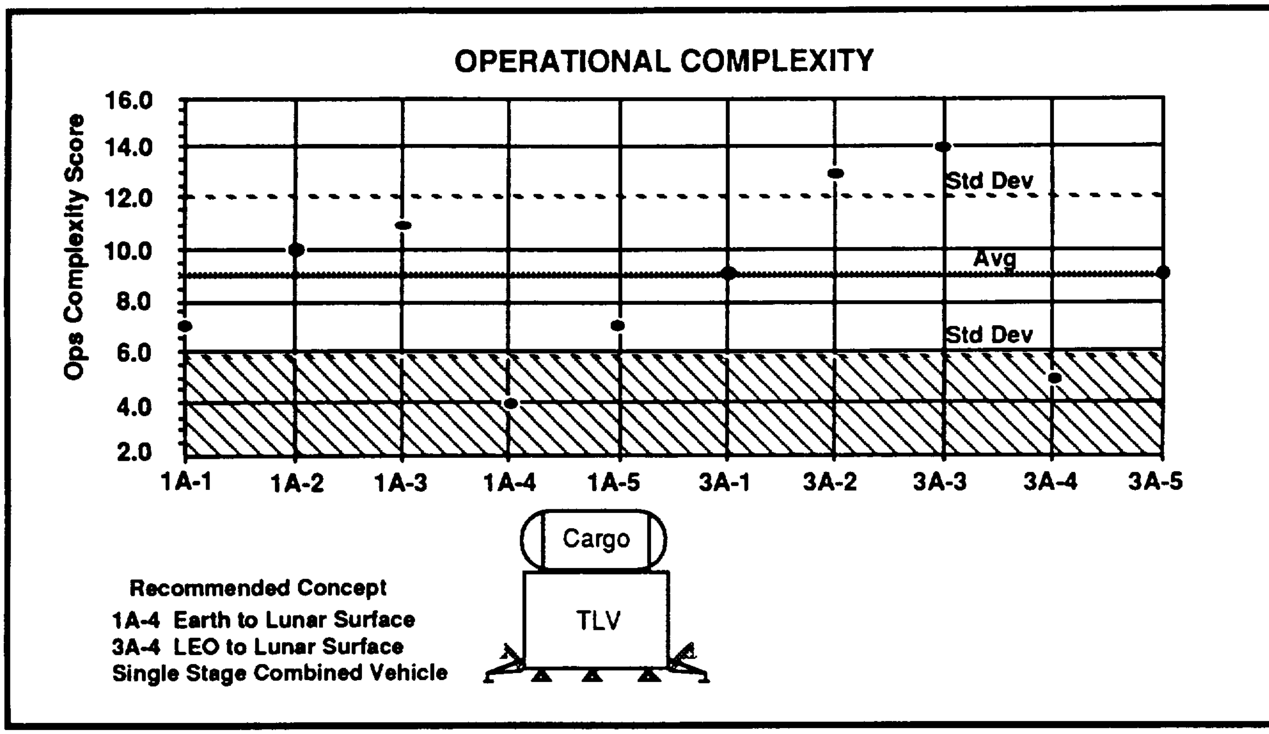


Figure 2.3.3.3-17 Cargo Only - Ops Complexity Summary

combined vehicle with drop tanks. Three concepts go from Earth direct to the lunar surface while two go from LEO to the lunar surface.

Summaries of the concept criteria data, operational complexity, performance, and relative cost, for the crew/cargo return concepts are illustrated in Figures 2.3.3.3-21 through -23. The average criteria scores for these concepts along with the one sigma standard deviations are plotted on each chart. Those concepts that fall below the standard deviation were recommended for additional review and study. For operational complexity, eight concepts (mostly single stage combined vehicles) were downselected for further study. Against the performance criteria, eight concepts (mostly separate or multistage vehicles with two crew cabs) were downselected for additional study. For the final criteria of relative cost, seven concepts (mostly single combined vehicles) were downselected for more study.

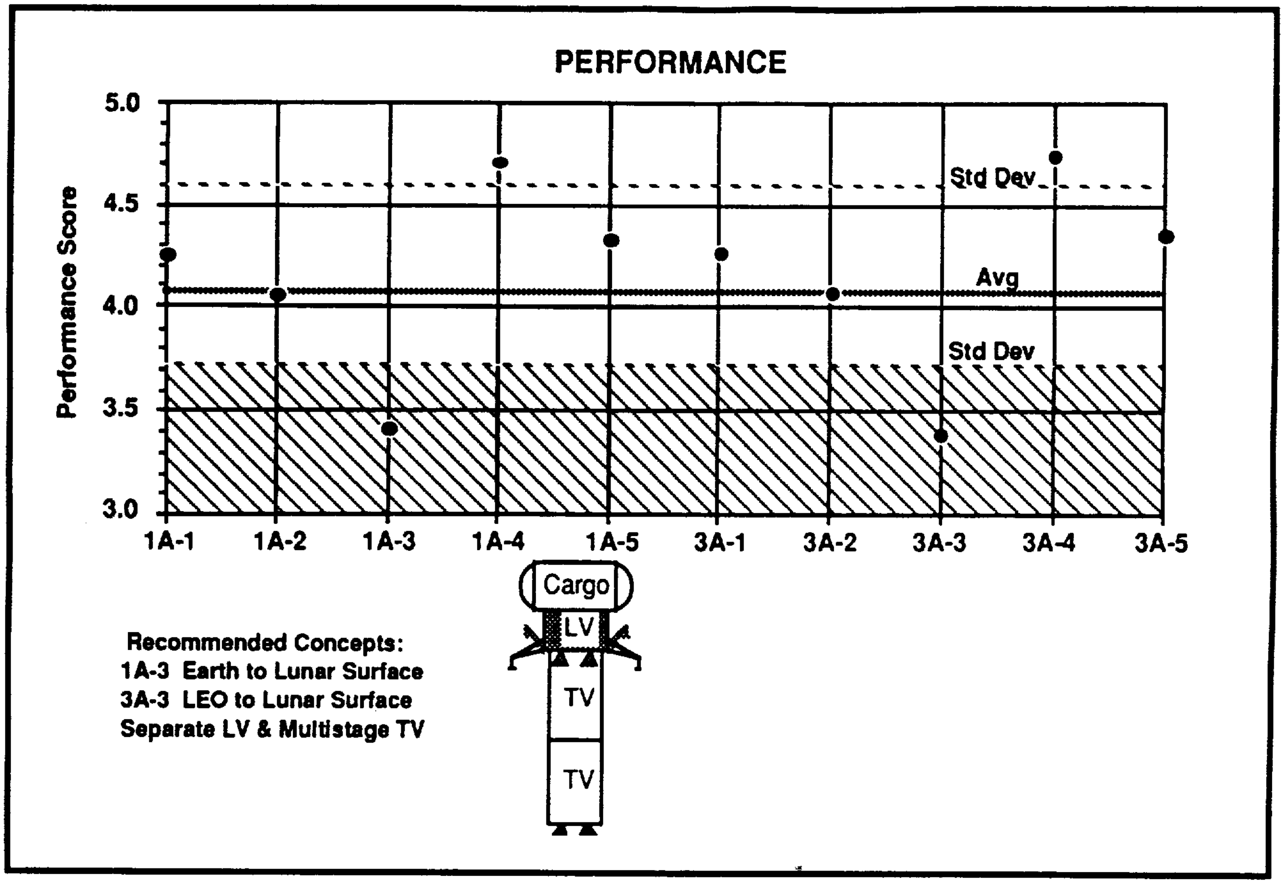


Figure 2.3.3.3-18 Cargo Only - Performance Summary

The recommended concepts for the crew/cargo scenario are shown in Figures 2.3.3.3-20 and -21. Concepts use separate single stage transfer and landing vehicles, single combined vehicles, multistage vehicles, and vehicles with drop tanks. Both aerobrake and ballistic return concepts were downselected, as well as single and dual crew cab concepts.

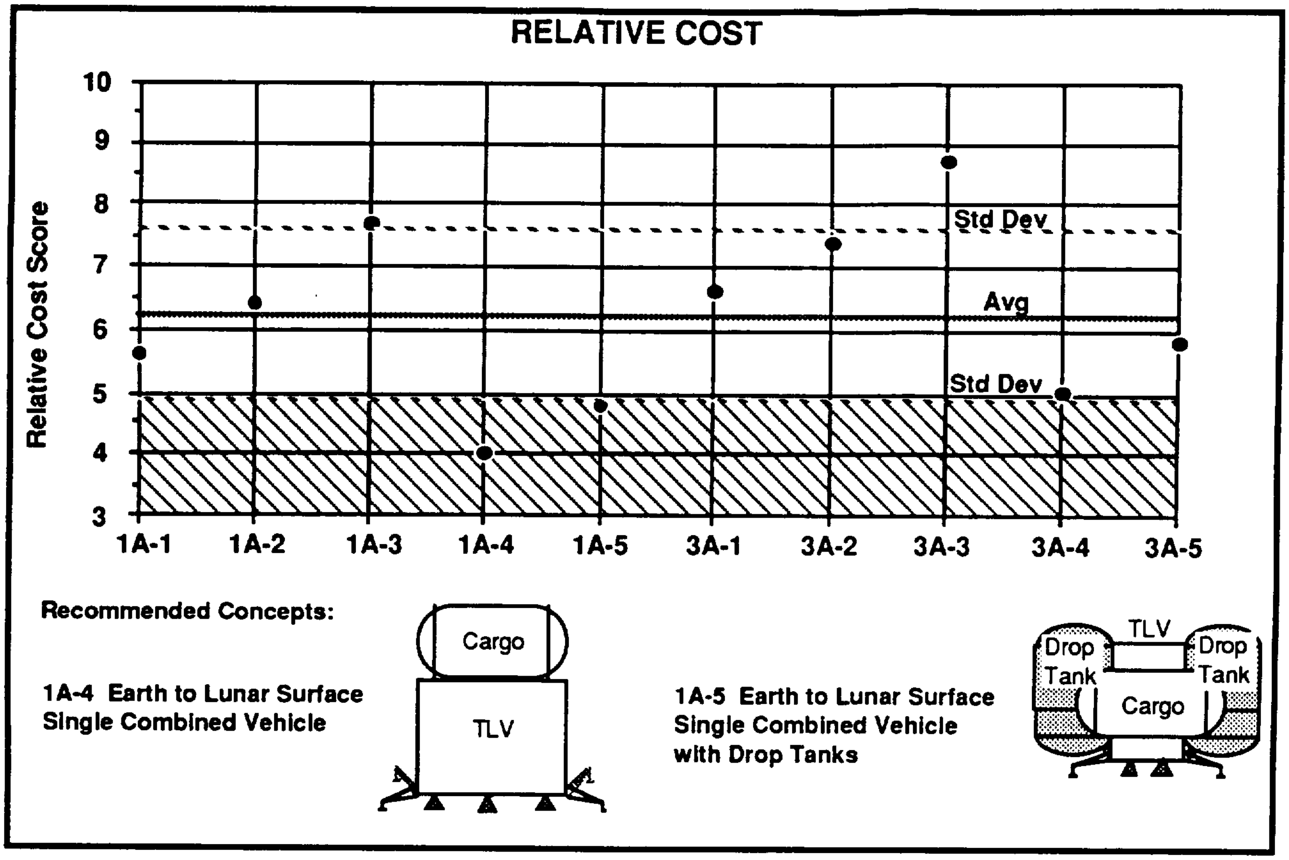


Figure 2.3.3.3-19 Cargo Only - Relative Cost Summary

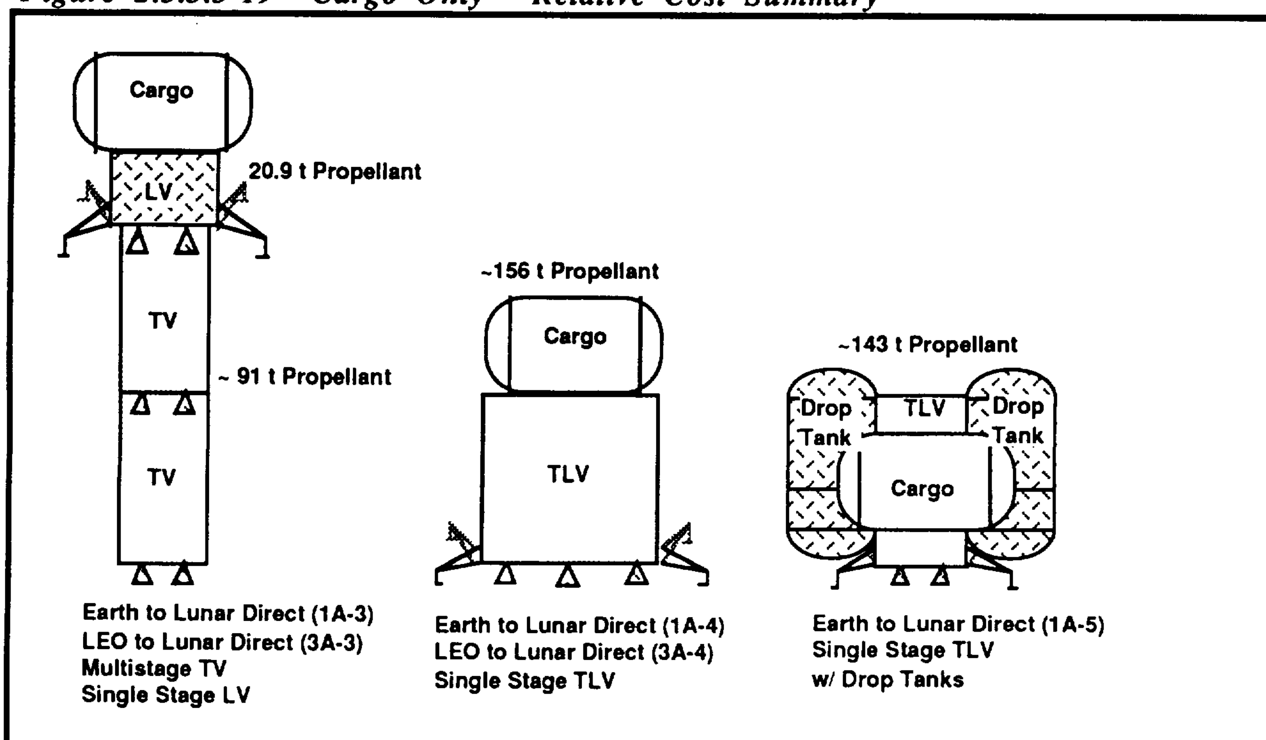


Figure 2.3.3.3-20 Cargo Only - Recommended Concepts

The concepts downselected for additional study and evaluation were highlighted on the vehicle stage matrix (Figure 2.3.3.3-27). Five cargo only concepts were recommended along with nineteen crew/cargo concepts. At least one concept from each orbital mechanics solution and each vehicle stage was downselected for further study.

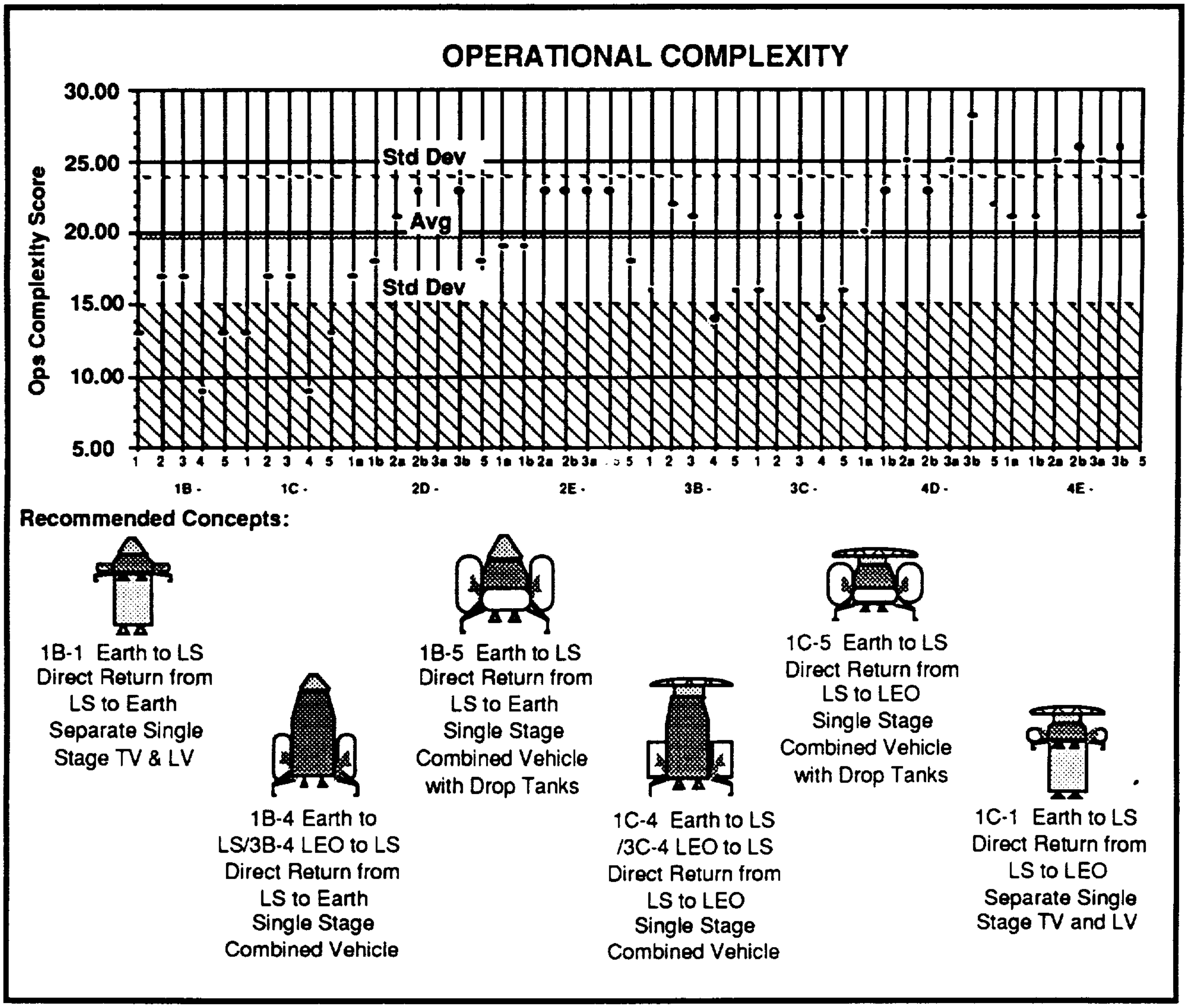


Figure 2.3.3.3-21 Crew/Cargo - Operations Complexity Summary

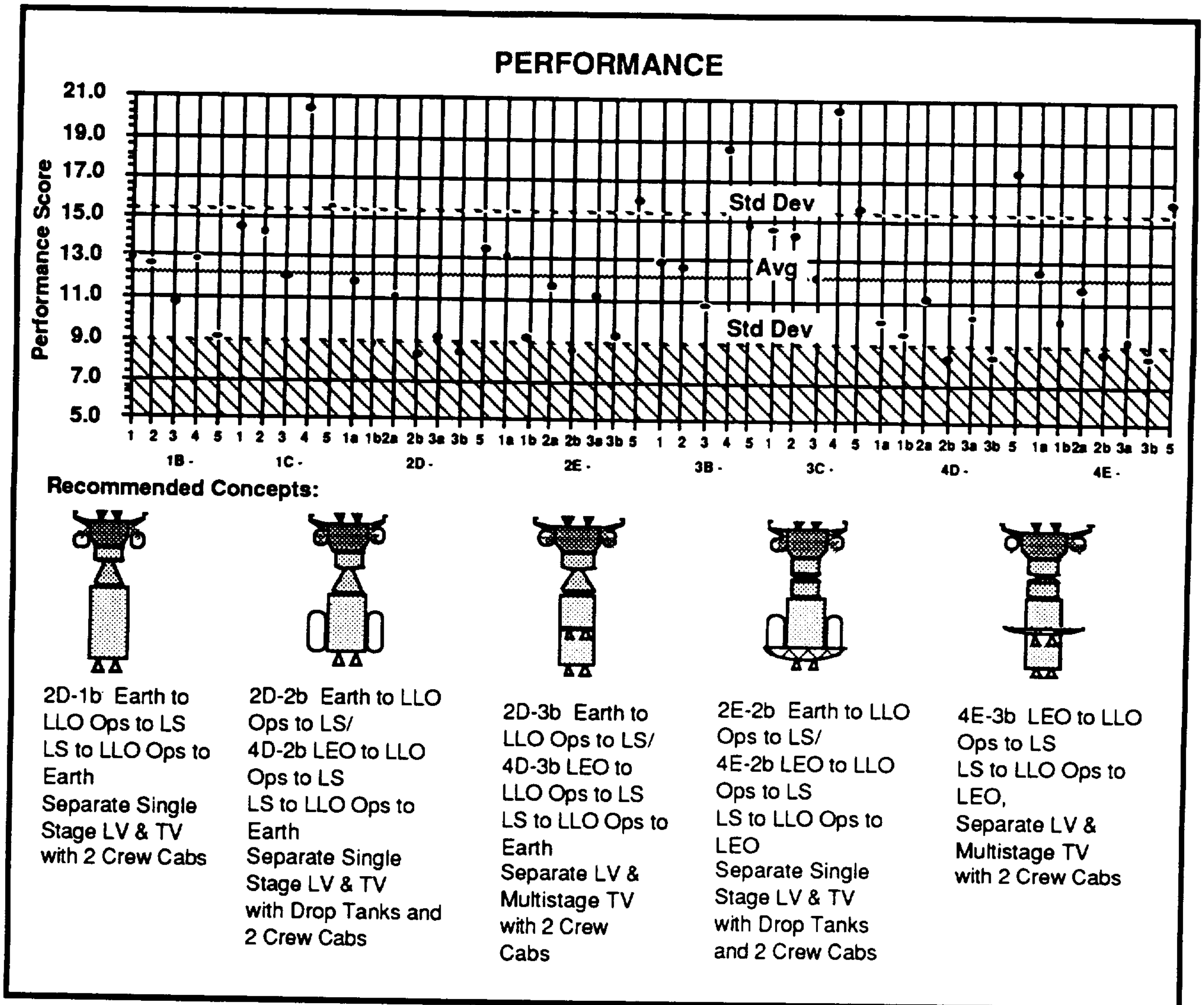
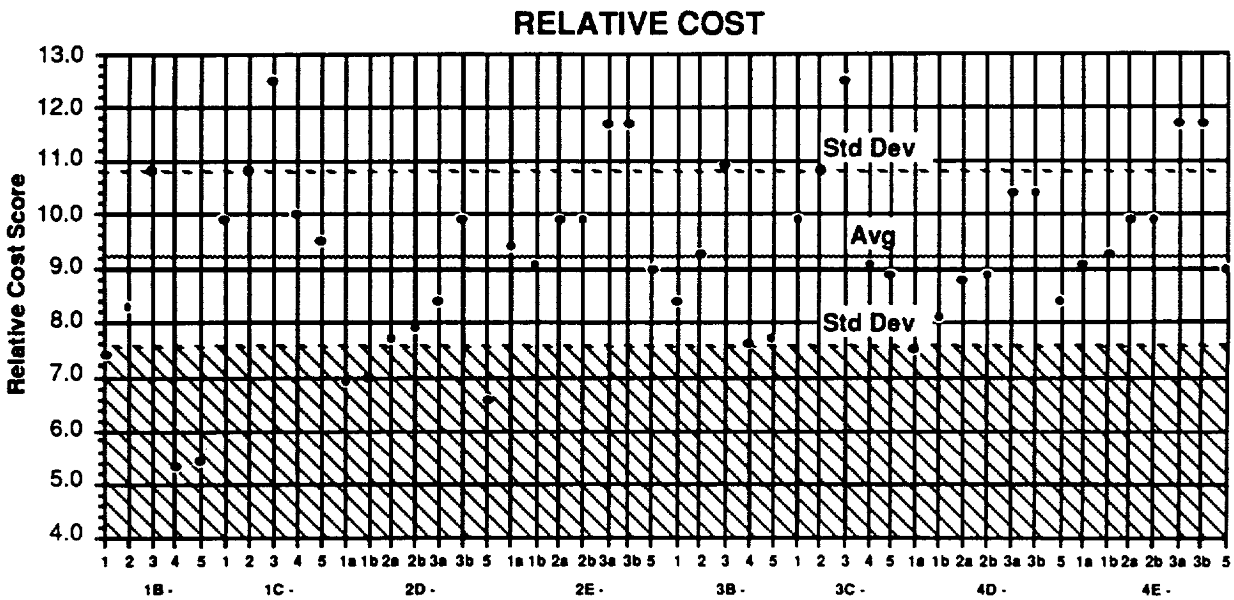


Figure 2.3.3.3-22 Crew/Cargo - Performance Summary



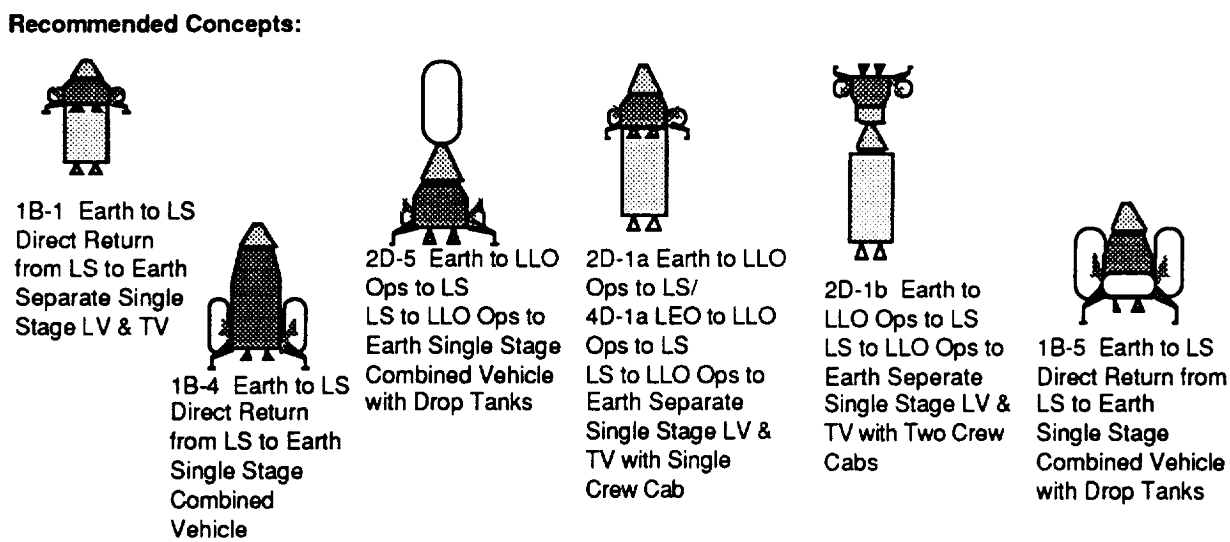


Figure 2.3.3.3-23 Crew/Cargo - Relative Cost Summary

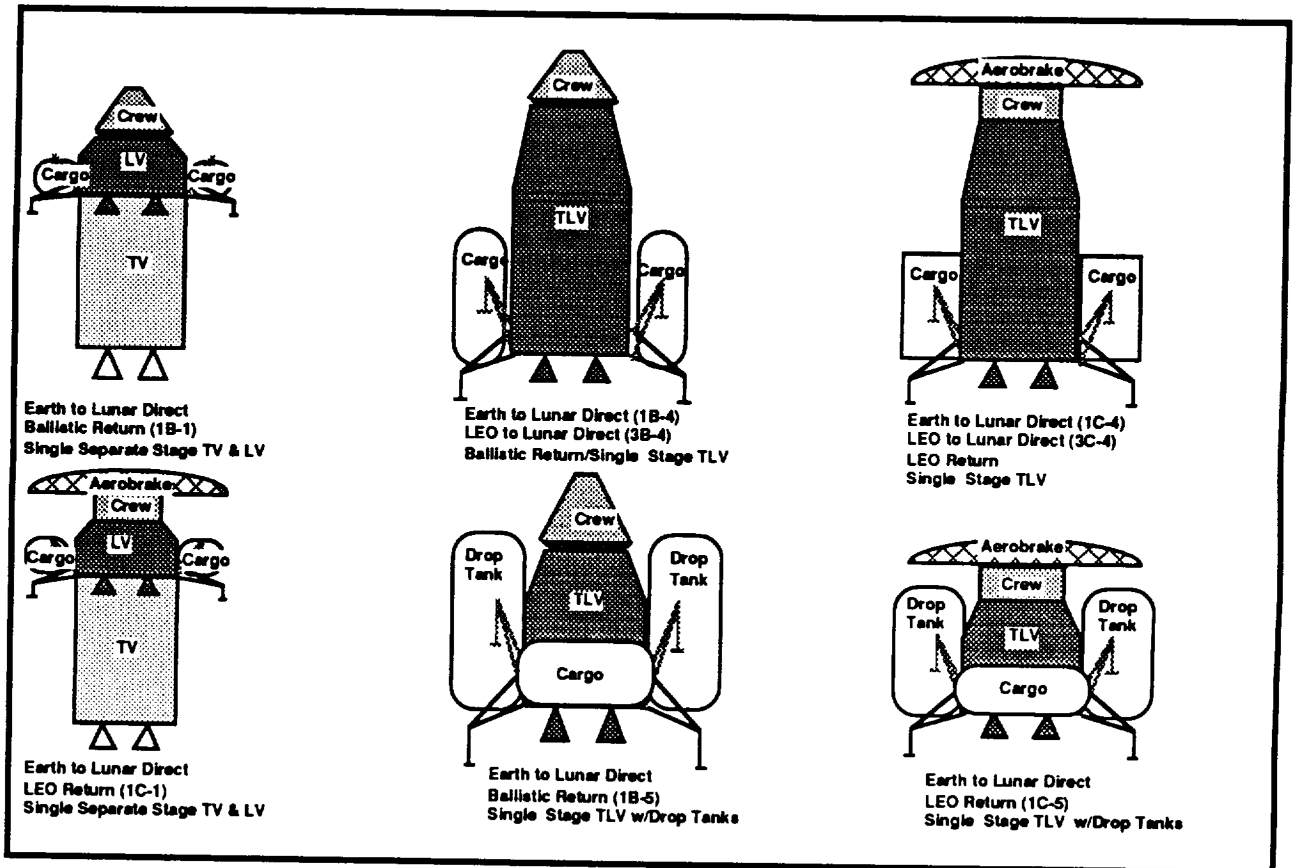


Figure 2.3.3.3-24 Crew/Cargo - Recommended Concepts

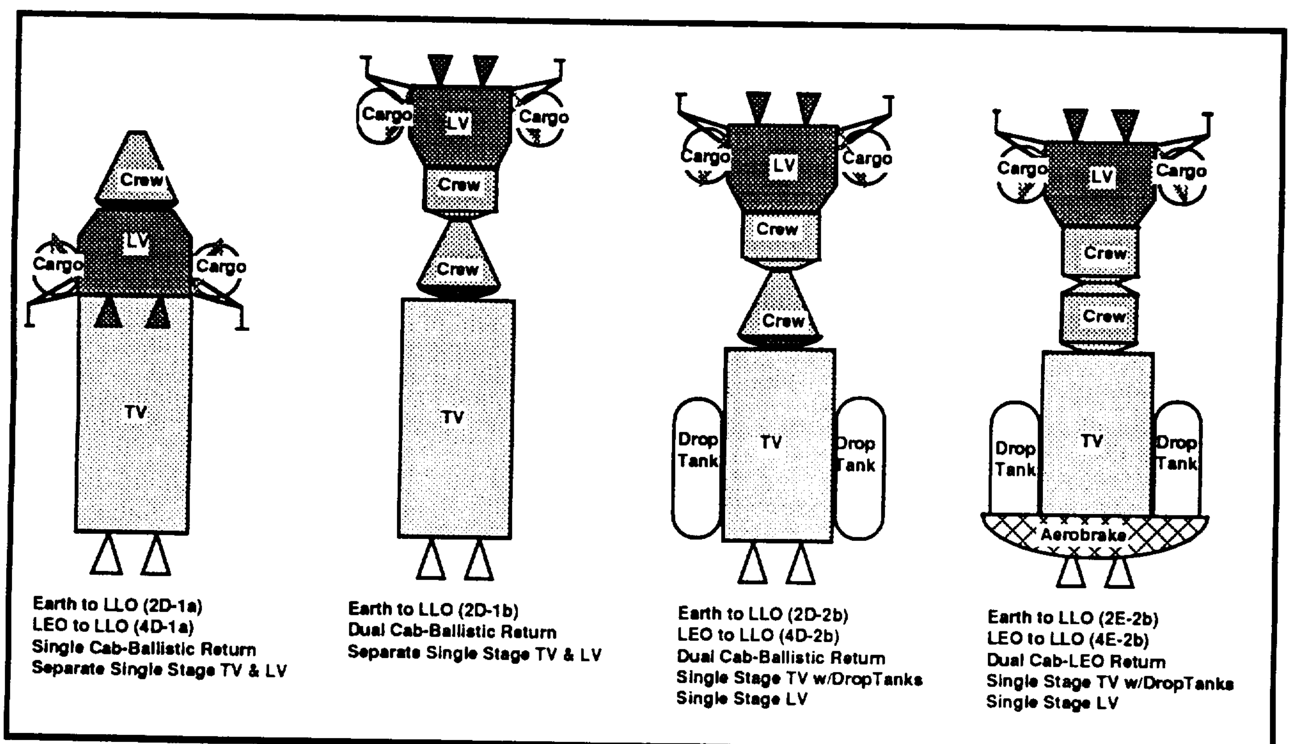


Figure 2.3.3.3-25 Crew/Cargo - Recommended Concepts

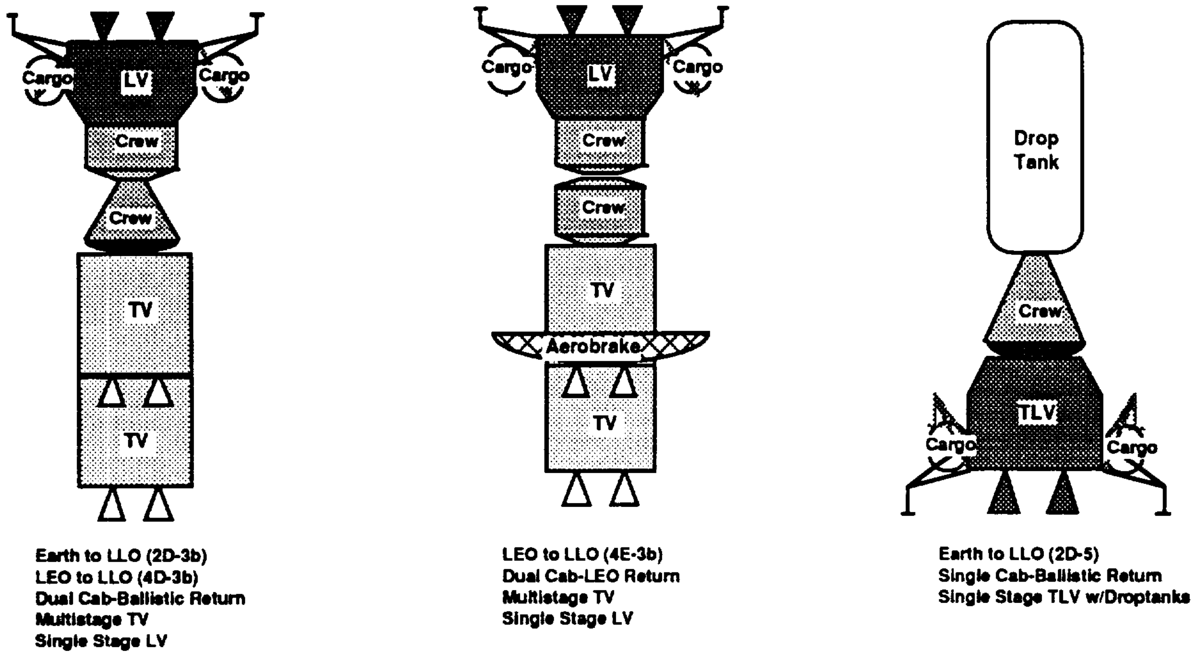


Figure 2.3.3.3.-26 Crew/Cargo - Recommended Concepts

		*	RETURN			
		A	B	C	D	E
LAUNCH PT	1	X	X	X		
	2				X	X
	3	X	X	X		
	4				X	X

* No Return

Launch Pt. - Up Leg From Earth / Leo

1:Earth To Lunar Surface
2:Earth To LLO Ops.. To Lunar Surface
3:LEO To Lunar Surface
4:LEO To LLO Ops.. To Lunar Surface

Return - Down Leg From Lunar Surface

A:No Return
B:Direct Return From Lunar Surface To Earth
C:Direct Return From Lunar Surface To LEO
D:From Lunar Surface To LLO Ops.. To Earth
E:From Lunar Surface To LLO Ops.. To LEO

		Separate Transfer Vehicle Separate Landing Vehicle (Multiple Propulsion Systems)			Combined Transfer & Landing Vehicle (Single Propulsion System)	
		Single Stage TV & LV 1	Single Stage TV & LV w/ Drop Tanks 2	Multistage TV & LV 3	Single Stage Combined Vehicle 4	Single Stage Combined Vehicle w/ Drop Tks. 5
No Return From Lunar Surface	1A	X		X	X	X
	3A	X	X	X	X	X
Launch Point / Return	1B	X	X	X	X	X
	1C	X	X	X	X	X
	2D	X X	X X	X	N/A	X
	2E	X X	X X	X X	N/A	X
	3B	X	X	X		X
	3C	X	X	X		X
	4D	X X	X X	X X	N/A	X
	4E	X X	X X	X X	N/A	X

Note: Two X's Indicate Single and Dual Crew Cab Concepts

O Recommended Concepts

Figure 2.3.3.3-27 Updated Vehicle Stage Matrix

2.3.3.3.2 Second Downselect Process—

2.3.3.3.2.1 Methodology—The concept selection process established to systematically evaluate and downselect STV concepts into a single concept or family of concepts (Fig. 2.3.3.3-2) continued during the second downselect phase. A concept selection methodology for this phase was developed to evaluate lunar concepts and recommend criteria driven concepts for final downselect (Fig. 2.3.3.3-28). Lunar architectures were developed and concepts were allocated to a particular architecture. A preliminary screening was performed of concepts recommended from the first downselected architectures, some new concepts, and some concepts added back from the initial downselect. Twelve concepts — five cargo only and seven crew/cargo concepts went through detailed concept definition during the second downselect phase. These concepts were evaluated against certain concept selection criteria – cost, operations, mission adaptability, and risk. Five criteria driven concepts — two cargo and three crew concepts — were recommended for additional study during the final selection process.

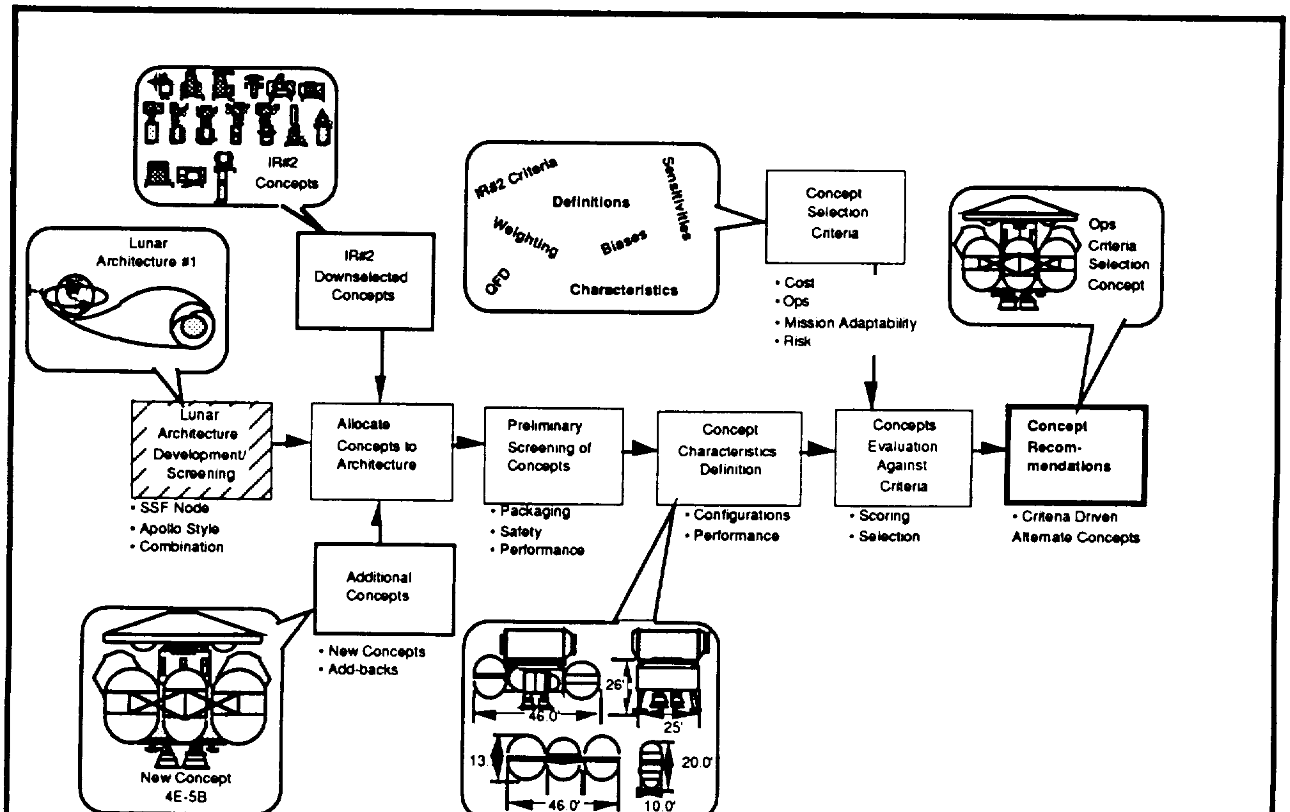


Figure 2.3.3.3-28 Concept Selection Methodology - Second Downselect

2.3.3.3.2.2 Groundrules and Assumptions—The following top level groundrules and assumptions were used in the second downselect process:

- Lunar DRMs were used to define the initial concepts.
- Downselected concepts were evaluated for adaptability to meet all DRMs.
- Cargo only expendable concepts were rated separately from crew and cargo missions.
- Cargo delivery requirements were 33 tonnes and crew/cargo delivery requirement was 14.6 tonnes.
- Subsystem definition was constant across all concepts, i.e., rigid aerobrake on all LEO return concepts, RL-10s on landing vehicles, etc.
- Initial flight and steady state flights were evaluated for all concepts.
- No constraints were placed on the Earth-to-Orbit (ETO) transportation system.
- All crew and cargo missions used LLO.

2.3.3.3.2.3 Lunar Architectures — The first step in the second downselect process was to develop lunar architectures and screen each lunar architecture against top level criteria such as LEO requirements and operations, technical risk, cost drivers, ground operations, etc. Five lunar architectures were developed. Option 1 uses LEO as the transportation node for both crew and cargo. Option 2 uses LEO as the transportation node for crew missions while cargo only missions proceed direct from Earth to the lunar surface. Option 3 is where both crew and cargo mission proceed direct from Earth to the lunar surface. Option 4 is where cargo and crew missions proceed direct from Earth to the lunar surface but crew missions return to LEO. Option 5 is where cargo missions proceed direct from Earth while crew missions proceed from LEO to the lunar surface but return to Earth. Details of the lunar architecture options are shown below:

- Option 1: Baseline LEO Transportation Node
  Cargo - from LEO to lunar surface
  Crew - from LEO to lunar surface, return to LEO
- Option 2: LEO Crew Node
  Cargo - from Earth to lunar surface
  Crew - from LEO to lunar surface, return to LEO
- Option 3: No Transportation Node (a la Apollo)
  Cargo - from Earth to lunar surface
  Crew - from Earth to lunar surface, return to Earth
- Option 4: LEO Crew Return Node
  Cargo - from Earth to lunar surface

Crew - from Earth to lunar surface, return to LEO

- Option 5: LEO Crew Node/Earth Return

Cargo - from Earth to lunar surface

Crew - from LEO to Lunar surface, return to Earth

A top level mission scenario of lunar architecture Option 1 is illustrated in Figure 2.3.3.3-29. Both the manned and cargo missions originate at the LEO transportation node. The manned return mission ends with rendezvous and docking with the LEO transportation node. In lunar architecture Option 2 (Fig. 2.3.3.3-30), the cargo missions begin with launch from Earth. Manned missions originate at the LEO transportation node and end with rendezvous and docking with the LEO transportation node. In Option 3 as shown in Figure 2.3.3.3-31, both manned and cargo missions begin with launch from Earth. Manned missions end with an Apollo style ballistic reentry and ocean landing. No LEO transportation node is required. Option 4 has both manned and cargo missions being launched from Earth with the crew return mission ending with rendezvous and docking at the LEO transportation node. In lunar architecture Option 5, the cargo missions begin with launch from Earth while the crew missions originate at the LEO transportation node but return Apollo style back to Earth.

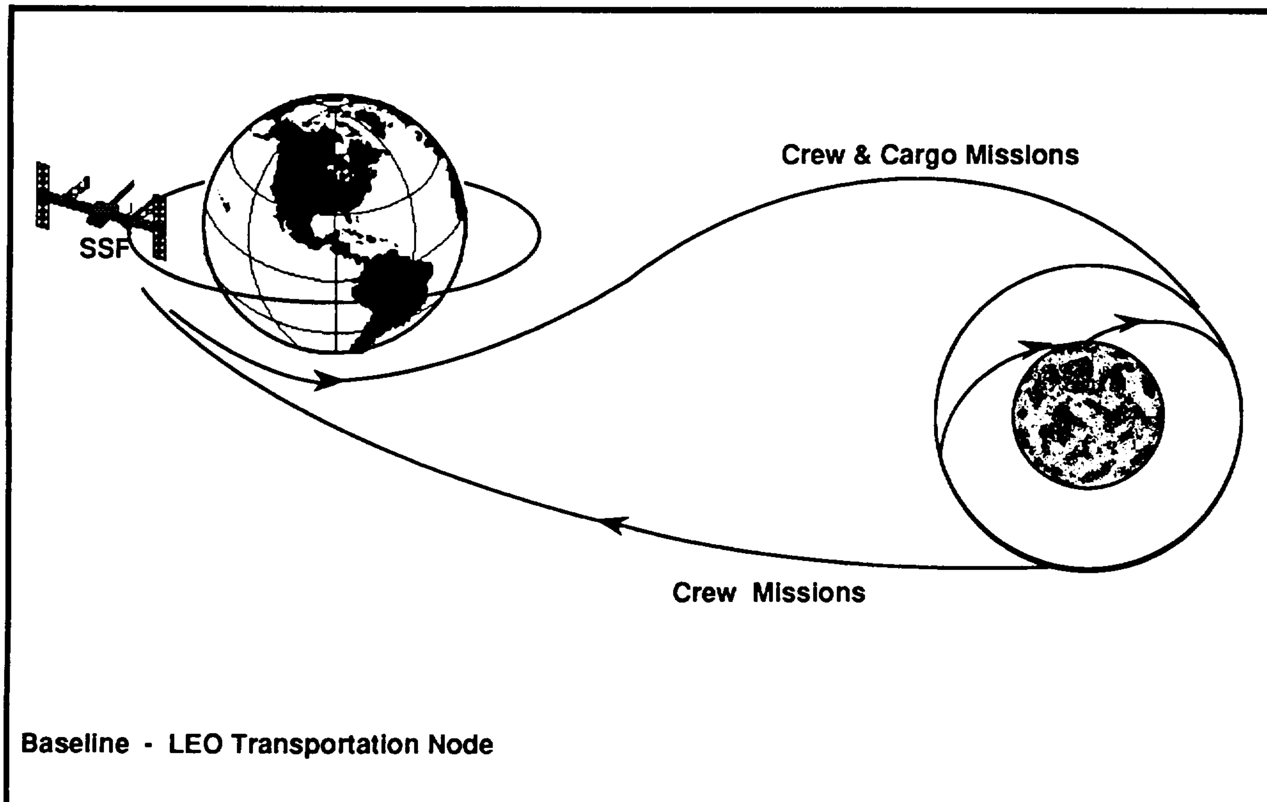


Figure 2.3.3.3-29 Lunar Architecture Option 1

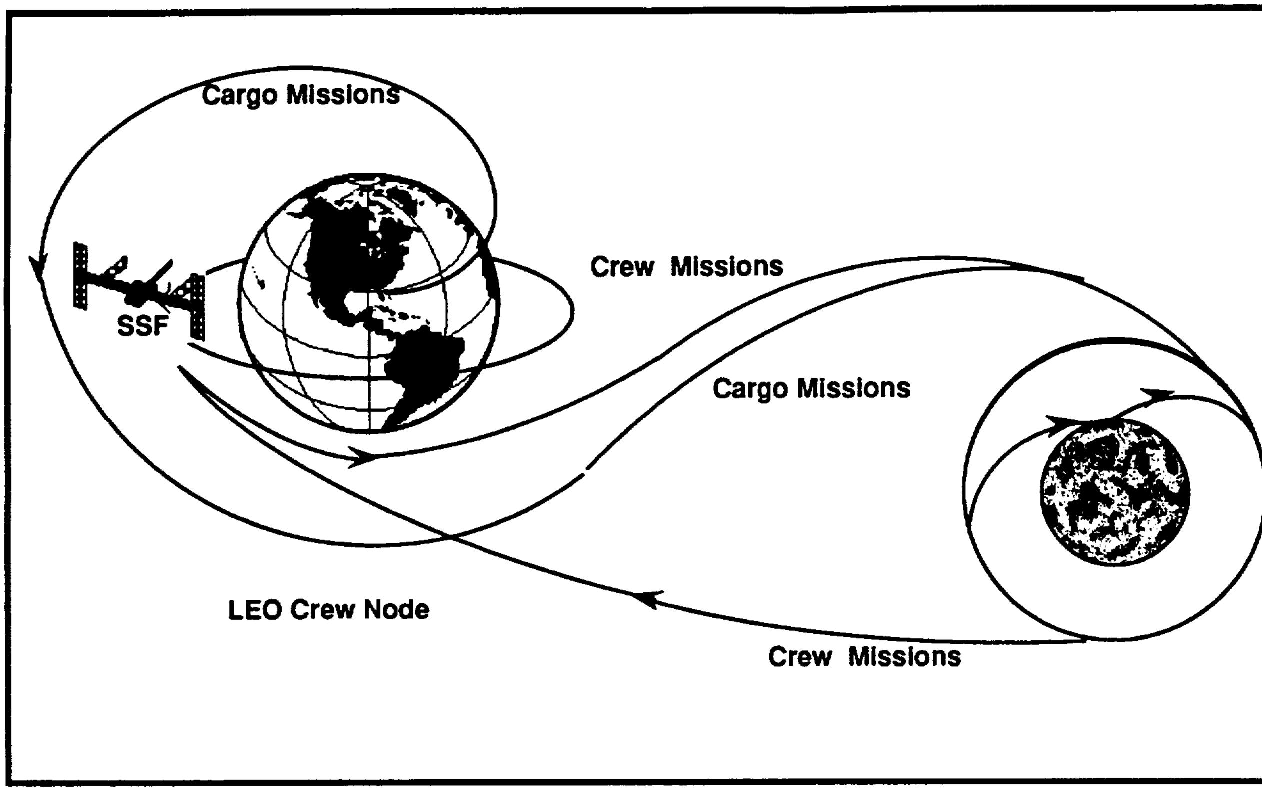


Figure 2.3.3.3-30 Lunar Architecture Option 2

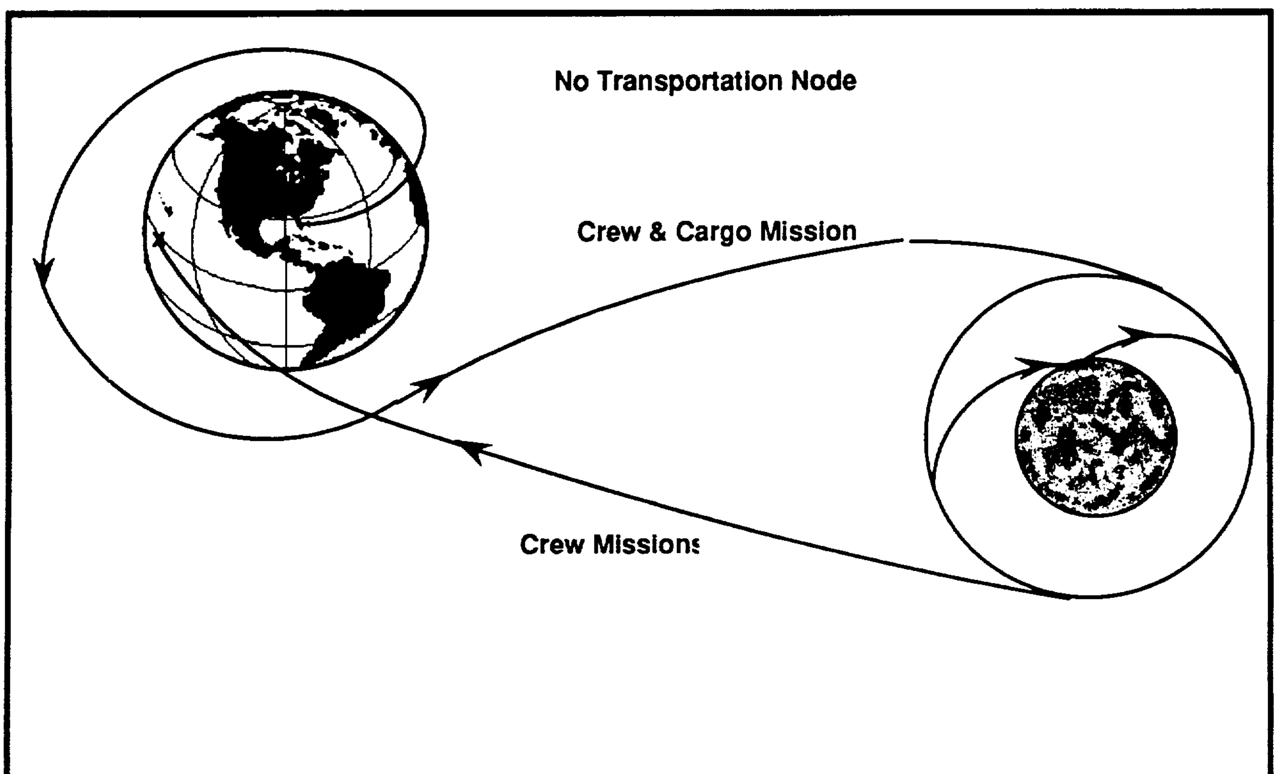
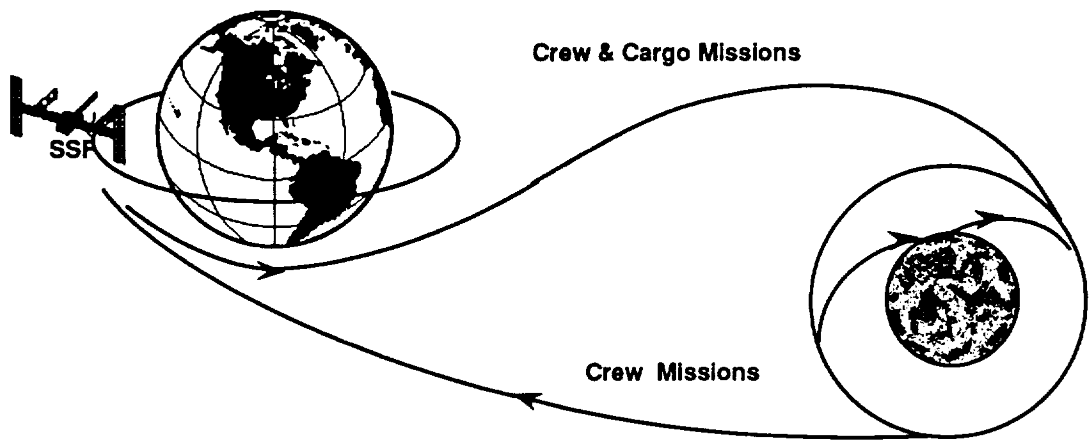


Figure 2.3.3.3-31 Lunar Architecture Option 3

C-2

Figure 2.3.3.3-32presents a top level assessment of these lunar architectures. Options 4 and 5 had the same impacts as Options 3 and 2, respectively. However, each of these options has the added inconvenience of the crew returning to a location other than its origin. This adds the impacts associated with transporting the crew and crew cab back to its origin for reuse. Therefore, lunar architectures 4 and 5 were discontinued from the study.



Two Preferred Options

Cargo	- LEO to Lunar Surface
Crew	- LEO to Lunar Surface, Return to LEO
Cargo	- Earth to Lunar Surface
Crew	- LEO to Lunar Surface, Return to LEO

<u>3: No Transportation Node (Apollo)</u>
Cargo - Earth to Lunar Surface
Crew - Earth to Lunar Surface, Return to Earth

<u>4: LEO Crew Return Node</u>
Cargo
Crew

<u>5: LEO Crew Node/Earth Return</u>
Cargo - Earth to Lunar Surface
Crew - LEO to Lunar Surface, Return to Earth

Figure 2.3.3.3-32 Lunar Architecture Assessment

2.3.3.3.2.4 Concept Allocation—Once the lunar architectures were defined, the next step was to allocate each concept to a particular lunar architecture. This allocation process included taking those concepts recommended for additional study as a result of the first downselect, generating some new concepts that were not evaluated before, and going back to reassess some concepts that were thrown out as a result of the first downselect. Figure 2.3.3.3-33 gives a brief description of cargo concepts that are applicable to lunar architecture Option 1 and a rationale for retention or deletion of the concepts from the study. Figures 2.3.3.3-34 & 35 give similar details for the cargo concepts for the other lunar architectures. A similar process was performed for the piloted concepts. Figures 2.3.3.3-36 through 2.3.3.3-38 show results for the allocation process for the piloted concepts against each of the three lunar architecture options.

Option 1 Cargo - LEO Transportation Node (LEO to LS/Return LEO)

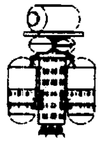
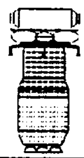
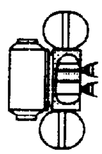
Concepts			Discussion	Result
3A-2		Single Stage Separate Transfer & Landing Vehicle with Drop Tanks	• Added back from IR#2 concepts • Separate elements make packaging/landing easier	Retained
3A-3		Multistage Transfer Vehicle and Separate Landing Vehicle	• Downselected from IR#2 concepts	Retained
3A-4		Single Propulsion Stage Combined Vehicle	• Downselected from IR#2 concepts • Large landing vehicle > 55'dia (control & stability uncertainty) • Cargo unloading on lunar surface - > 46' above LS	Deleted
3A-5		Single Propulsion Stage Combined Vehicle with Drop Tanks	• Added back from IR#2 concepts • Drop tanks make packaging/landing easier	Retained

Figure 2.3.3.3-33 Cargo Concepts Allocated to Lunar Architecture 1

Option 2 Cargo - LEO Crew Node (Crew From LEO to LS/Return LEO)

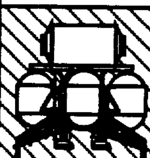
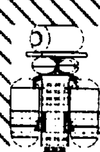
Concepts			Discussion	Result
1A-1		Single Stage Separate Transfer & Landing Vehicle with Drop Tanks	• Added back from IR#2 concepts • Separate elements make packaging/landing easier	Retained
1A-3		Multistage Transfer Vehicle and Separate Landing Vehicle	• Downselected from IR#2 concepts	Retained
1A-4		Single Propulsion Stage Combined Vehicle	• Downselected from IR#2 concepts • Large landing vehicle > 55'dia (control & stability uncertainty) • Cargo unloading on lunar surface - > 40' above LS	Deleted
1A-5		Single Propulsion Stage Combined Vehicle with Drop Tanks	• Downselected from IR#2 concepts • Packaging in Launch Vehicle Requires > 60' dia shroud	Deleted

Figure 2.3.3.3-34 Cargo Concepts Allocated to Lunar Architecture 2

Option 3 Cargo - No Transporation Node (Earth to LLO to LS/Return Earth)

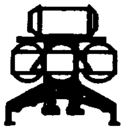
Concepts			Discussion	Result
1A-1		Single Stage Separate Transfer & Landing Vehicle with Drop Tanks	• Added back from IR#2 concepts • Separate elements make packaging/landing easier	Retained
1A-3		Multistage Transfer Vehicle and Separate Landing Vehicle	• Downselected from IR#2 concepts	Retained
1A-4		Single Propulsion Stage Combined Vehicle	• Downselected from IR#2 concepts • Large landing vehicle > 55'dia (control & stability uncertainty) • Cargo unloading on lunar surface > 40' above LS	Deleted
1A-5		Single Propulsion Stage Combined Vehicle with Drop Tanks	• Downselected from IR#2 concepts • Packaging in Launch Vehicle Requires Shroud - > 60' dia	Deleted

Figure 2.3.3.3-35 Cargo Concepts Allocated to Lunar Architecture 3

Option 1 Piloted - LEO Transportation Node (LEO to LS/Return LEO)

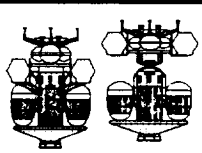
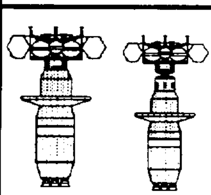
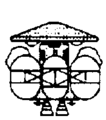
Concepts			Discussion	Result
3C-4		Single Propulsion Stage Combined Vehicle	• Downselected from IR#2 concepts • No LLO ops for crew missions violates safety reqts	Deleted
4E-2A/4E-2B		Single Stage Separate Transfer & Landing Vehicle with Drop Tanks & Single Crew Cab/Dual Crew Cabs	• Single crew cab added back from IR#2 concepts. Dual crew cab downselected from IR#2. • 90-Day Baseline Reference	Retained
4E-3A/4E-3B		Multistage Transfer Vehicle & Separate Landing Vehicle Single Crew Cab/Dual Crew Cabs	• Single crew cab added back from IR#2 concepts. Dual crew cab downselected from IR#2.	Retained
4E-5B		Single Propulsion Stage Combined Vehicle with Drop Tanks	• Added as new concept • No aerobrake penetrations, returns landing vehicle	Retained

Figure 2.3.3.3-36 Piloted Concepts Allocated to Lunar Architecture 1

Option 2 Piloted - LEO Crew Node (Crew From LEO to LS/Return LEO)

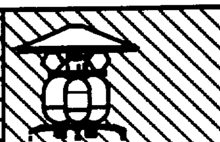
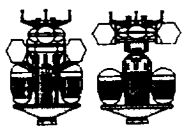
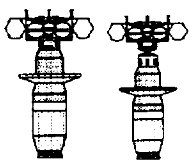
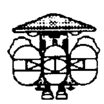
Concepts			Discussion	Result
3C-4		Single Propulsion Stage Combined Vehicle	• Downselected from IR#2 concepts • No LLO ops for crew missions violates safety reqts	Deleted
4E-2A/4E-2B		Single Stage Separate Transfer & Landing Vehicle with Drop Tanks & Single Crew Cab/Dual Crew Cabs	• Single crew cab added back from IR#2 concepts. Dual crew cab downselected from IR#2 concepts. • 90-Day Baseline Reference	Retained
4E-3A/4E-3B		Multistage Transfer Vehicle & Separate Landing Vehicle Single Crew Cab/Dual Crew Cabs	• Single crew cab added back from IR#2 concepts. Dual crew cab downselected from IR#2.	Retained
4E-5B		Single Propulsion Stage Combined Vehicle with Drop Tanks	• Added as new concept • No aerobrake penetrations, returns landing vehicle • Good performance	Retained

Figure 2.3.3.3-37 Piloted Concepts Allocated to Lunar Architecture 2

Option 3 Piloted - No Transportation Node (Earth to LLO to LS/Return Earth)

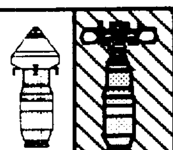
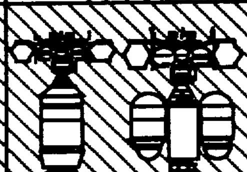
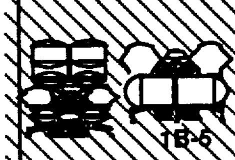
Concepts			Discussion	Result
1B-1/2D-1A		Single Stage Separate Transfer Vehicle & Landing Vehicle	• Downselected from IR#2 concepts • No LLO Ops for crew missions violates safety Req'ts (1B-1)	1B-1 Deleted 2D-1A Retained
2d-3A/B		Multistage Transfer vehicle & Separate landing vehicle 1 & 2 Crew Cabs	• 2D-3B Downselected, Single Cab 2D-3A Added back from IR#2. • Safety problem for packaging 2 crew cabs inside launch vehicle	2D-3a Retained 2D-3b Deleted
2D-1B/2D-2B		Single Stage Separate Transfer & Landing Vehicles Single/Dual Crew Cabs	• Downselected from IR#2 concepts • Dual Crew Pose Safety Problems, Crew Inside Launch Vehicle Shroud - safety issue for abort.	Deleted
1B-4/1B-5/2D-5		Single Propulsion Stage Combined Vehicle & with Drop Tanks	• Downselected From IR#2 Concepts • No LLO Ops Violates Safety Requirements	Deleted

Figure 2.3.3.3-38 Piloted Concepts Allocated to Lunar Architecture 3

After the concept allocation and top level screening process, five cargo concepts, 1A-1, 1A-3, 3A-2, 3A-3, and 3A-5 (Fig. 2.3.3.3-39), were retained for additional study and definition. Five crew concepts, (4E-2A, 4E-2B, 4E-3A, 4E-3B, and 4E-5B), shown in Figure 2.3.3.3-40, were retained after the preliminary screening for lunar architectures Options 1 & 2. Two crew concepts, 2D-1A and 2D-3A (Figure 2.3.3.3-41), were retained after the preliminary screening for lunar architecture Option 3.

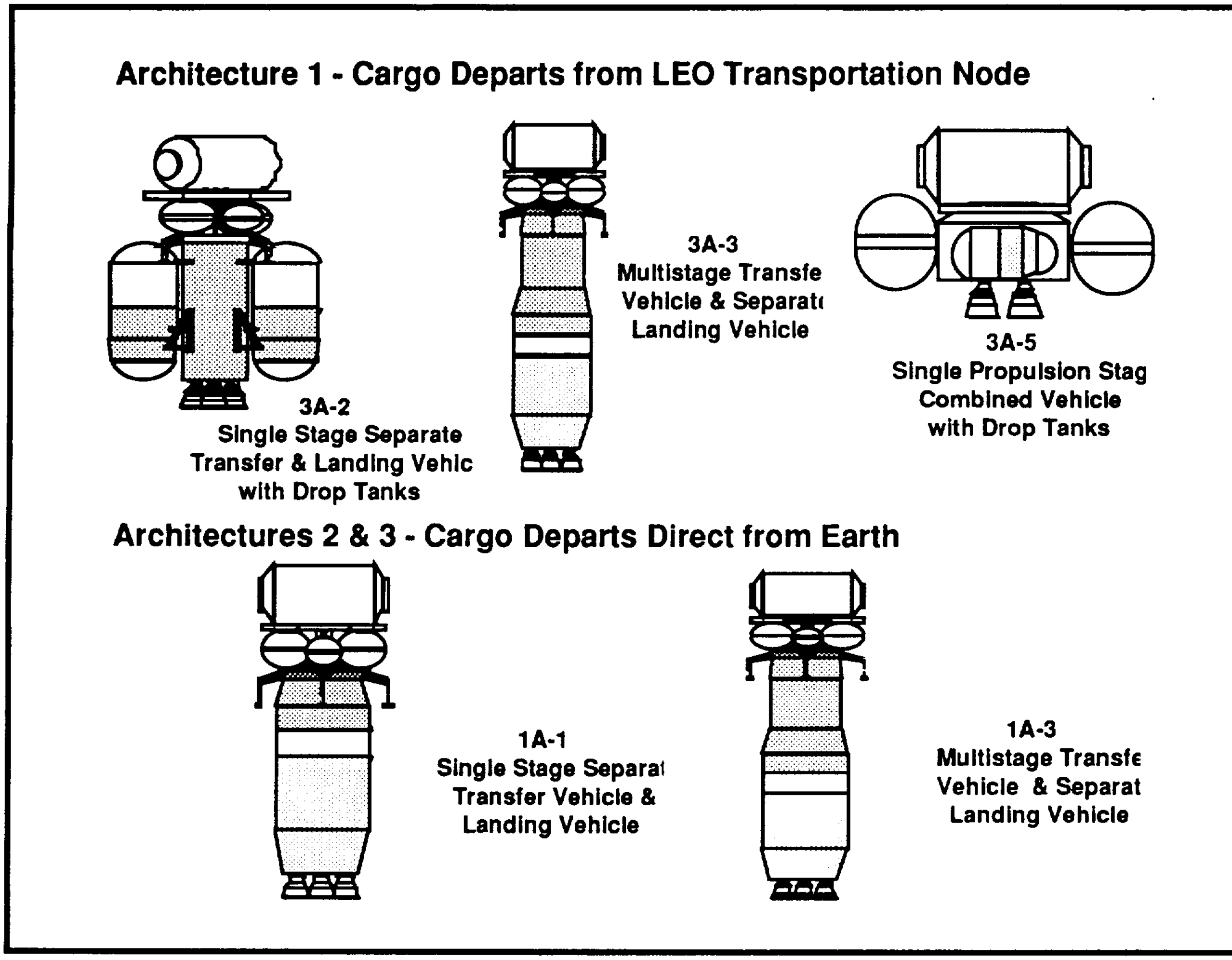


Figure 2.3.3.3-39 Cargo Concepts Retained for Additional Study

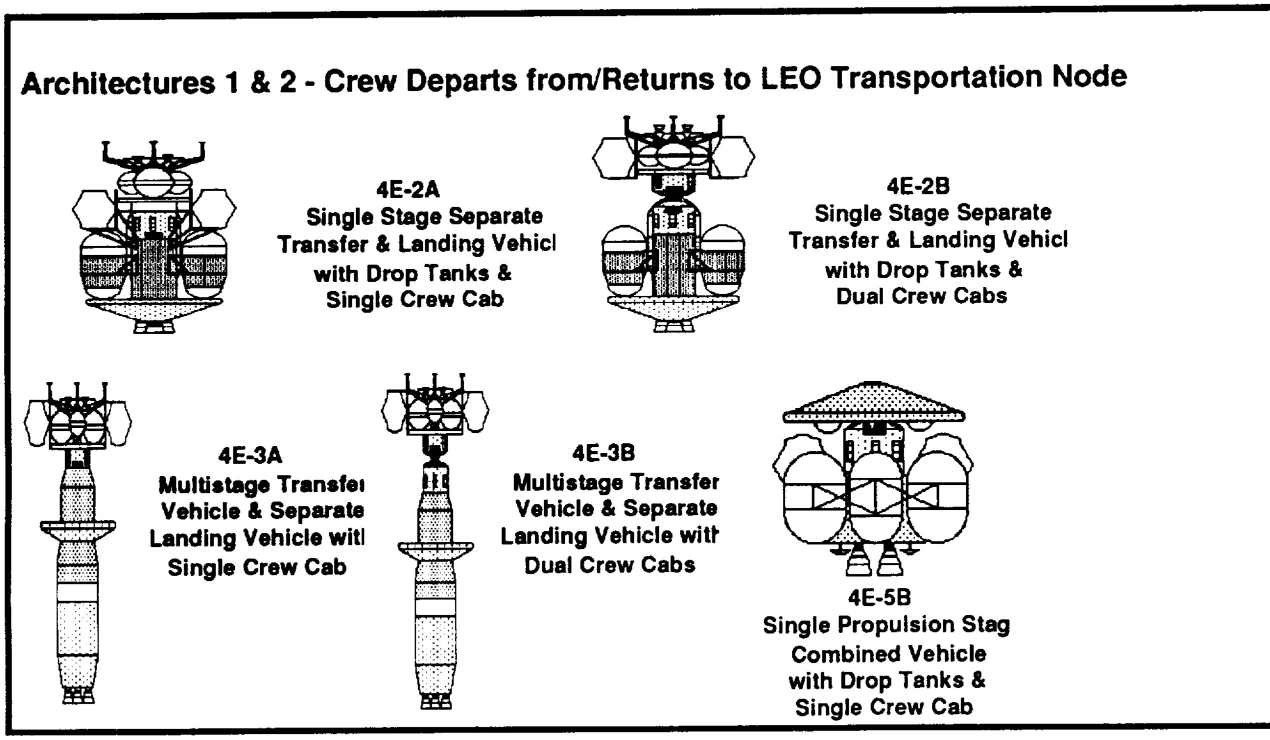


Figure 2.3.3.3-40 Piloted Concepts Retained for Additional Study

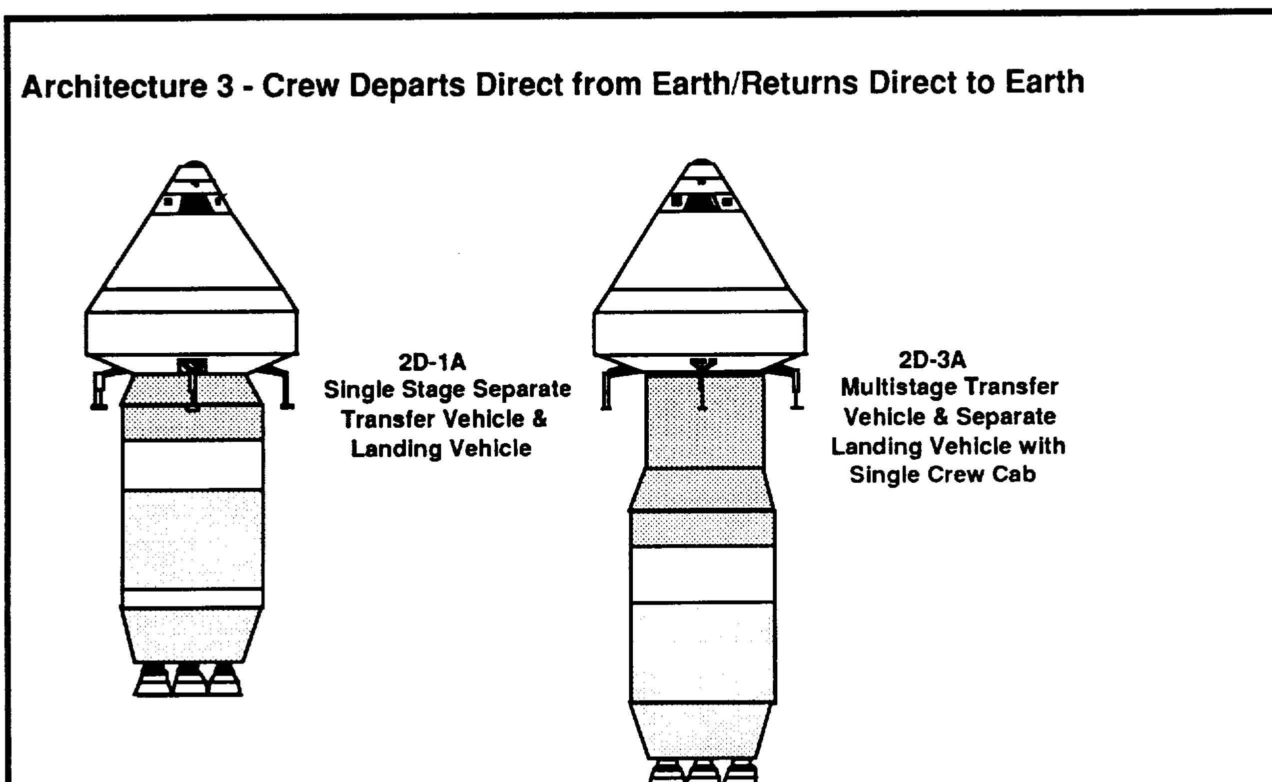


Figure 2.3.3.3-41 Piloted Concepts Retained for Additional Study

2.3.3.3.2.5 Detailed Concept Definition—The five cargo and seven crew concepts that passed the initial screening were then subjected to more detailed concept definition. Top level mission scenarios (outbound and inbound legs) were generated for each concept. An assessment of critical mission operations during each mission phase was evaluated for criticality 1 (loss of crew or mission critical hardware) and criticality 2 operations (loss of mission - crew returns safely, cargo can be salvaged). Detailed configuration definitions for each concept were developed that included preliminary sizing, dimensions, and mass properties. In addition, manifest layouts were generated for each concept to show typical flight manifesting in HLLVs. Each concept's ability to adapt to other design reference missions was assessed by addressing vehicle element interchange ability and performance capability. Operational timelines were generated for each concept to determine workshifts required at SSF for the initial vehicle assembly and steady state refurbishment operations. New ground operations facilities for each concept were also determined. Cost data for each concept was broken up into DDT&E, production, operations, and total LCC by vehicle element. Figures 2.3.3.3-42 through 2.3.3.3-46 illustrate the typical data generated for each concept (crew concept 4E-5B is shown as an example). Appendix 2 contains all the detailed configuration data for the five cargo and seven crew concepts.

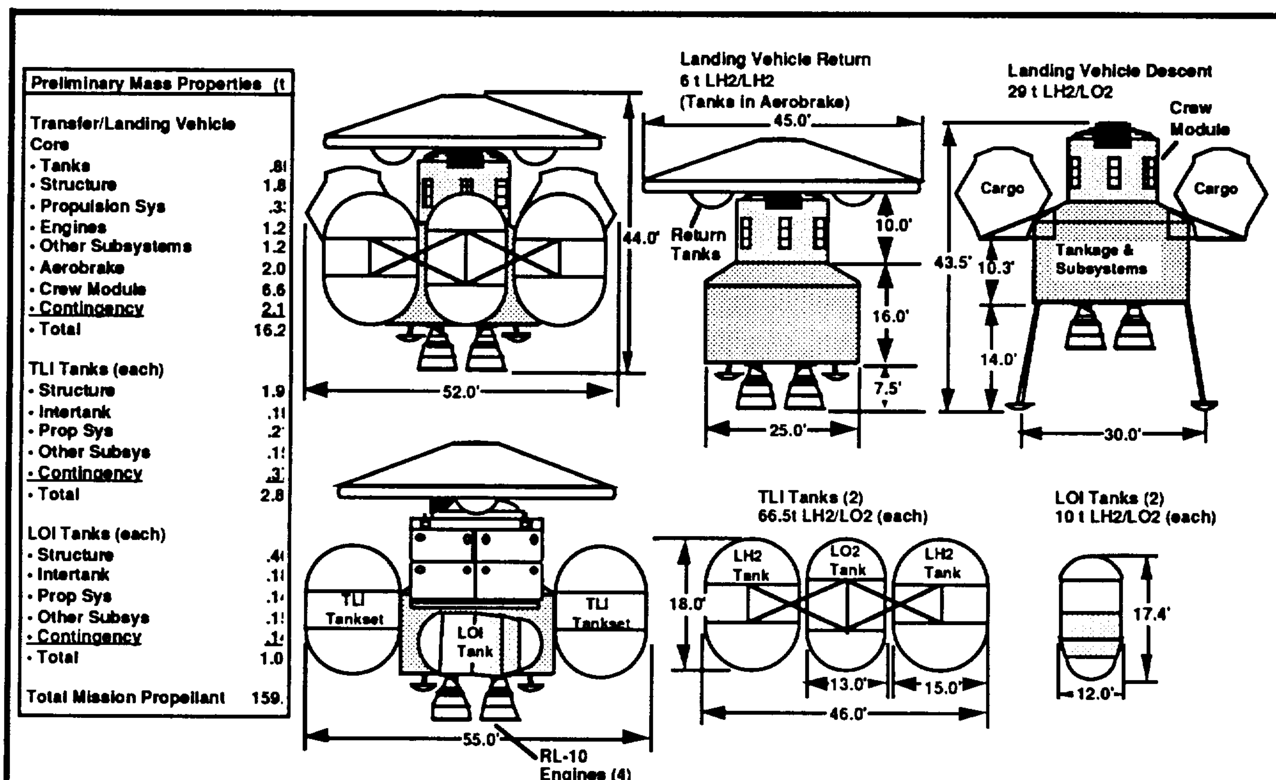


Figure 2.3.3.3-42 Typical Detail Data - Configuration Definition

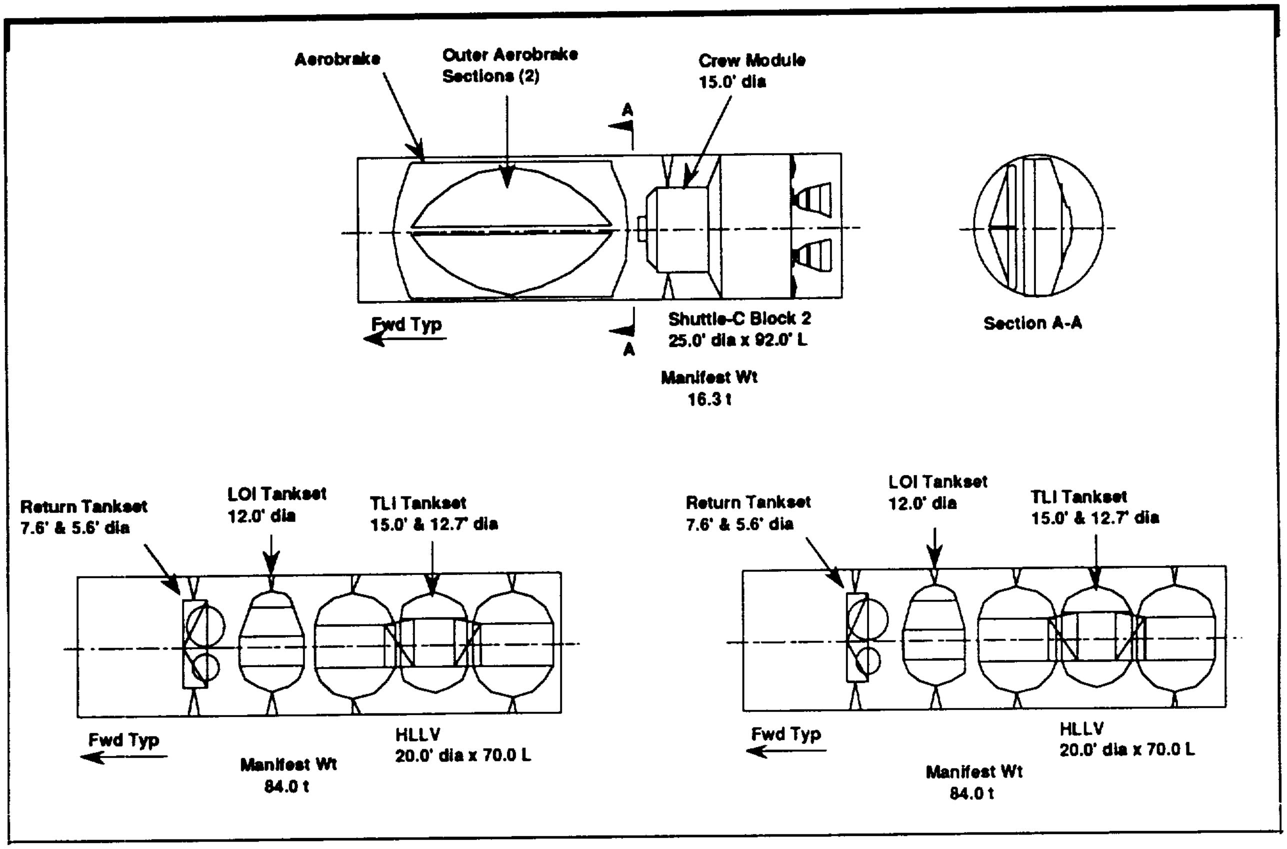


Figure 2.3.3.3-43 Typical Detail Data - Manifest Layout

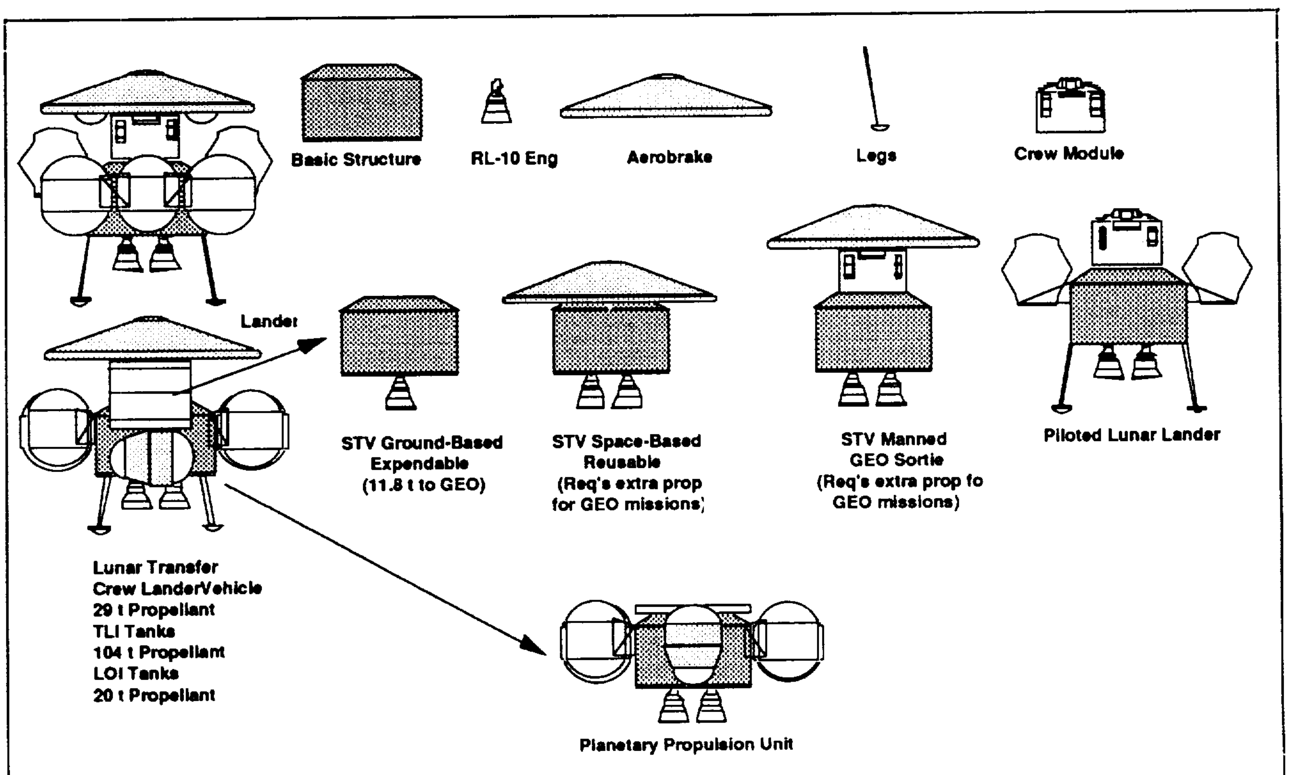
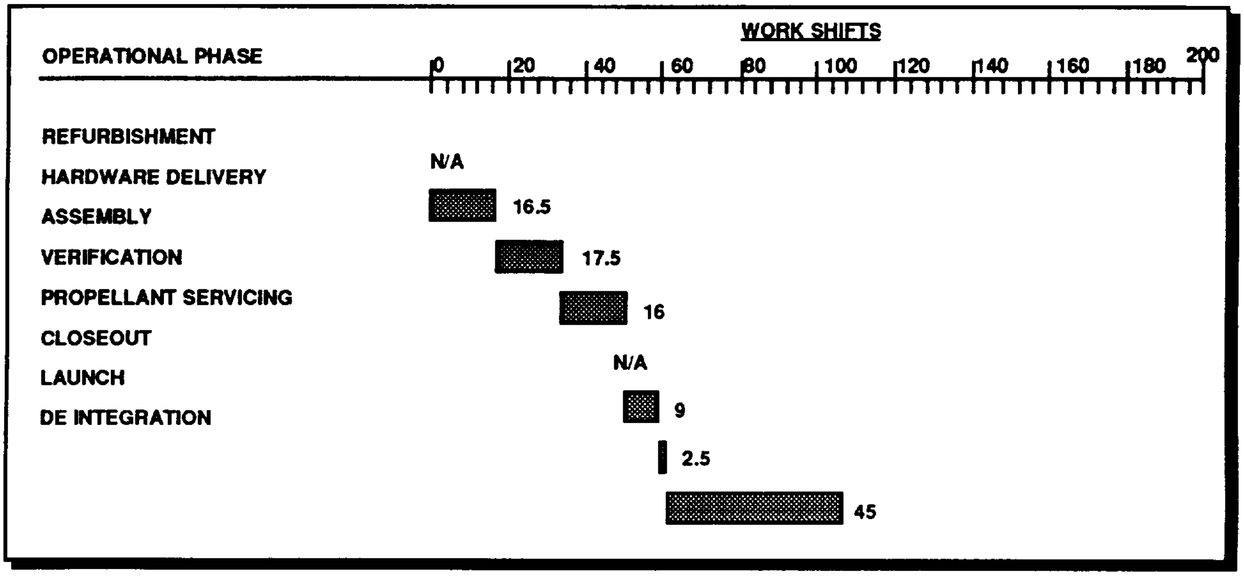


Figure 2.3.3.3-44 Typical Detail Data - Adaptability to Other Missions



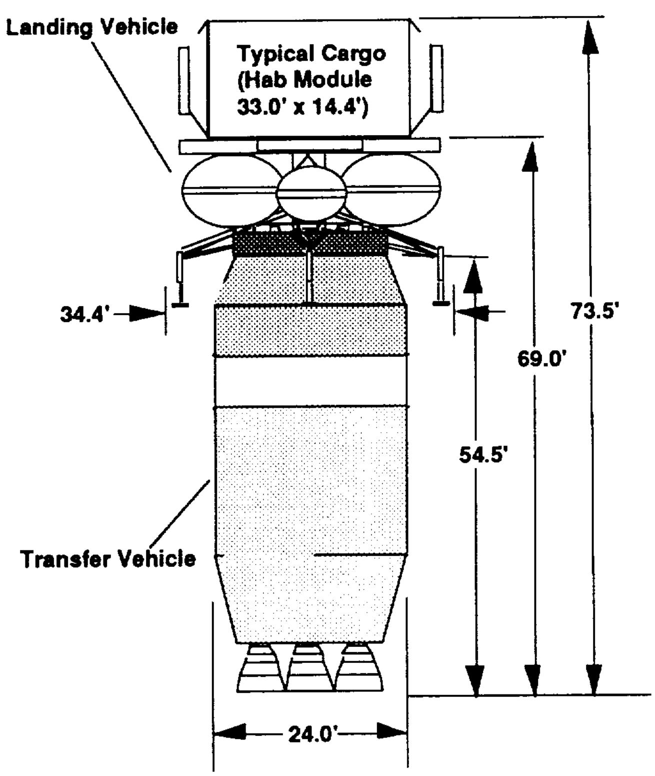
- **Lowest of initial manned concepts in shifts and complexity due primarily to fully assembled TLV and simplified Aerobrake.**

Figure 2.3.3.3-45 Typical Detail Data - Operational Timelines

Element	DDT&E	Prod	Ops	LCC
Core Stage/Lander	1113.5	1547.6	0.0	2661.1
TLI Tanks	74.7	462.3	0.0	537.0
LOI Tanks	42.9	255.8	0.0	298.7
Software	500.0	0.0	0.0	500.0
Support Equipment	492.5	0.0	0.0	492.5
System Test	1634.1	0.0	0.0	1634.1
Facilities	2484.4	0.0	0.0	2484.4
Operations	60.0	0.0	100.0	160.0
Systems Engineering	989.1	339.9	15.0	1344.0
Program Management	739.1	260.6	11.5	1011.2
ETO Costs	0.0	0.0	19,026.4	19,026.4
LEO Node Costs	2000.0	0.0	0.0	2000.0
Payload Costs	0.0	0.0	0.0	0.0
TOTAL	10,130.3	2866.2	19,152.9	32,149.4

Figure 2.3.3.3-46 Typical Detail Data - Cost Data

For each of the twelve concepts, a configuration summary sheet was generated that provides key features, preliminary mass properties, and overall concept configuration and dimensions. Figure 2.3.3.3-47 illustrates the summary sheet for cargo concept 1A-1. This is a single stage transfer and separate landing vehicle launched completely assembled in an HLLV direct from Earth to the lunar surface. The HLLV requires a 36.0' diameter shroud and a lift capability of over 180 t (147.4 t vehicles and propellant plus 33 t cargo). The transfer vehicle has 4 RL-10 engines and requires a propellant capacity of 113.9 t. The landing vehicle has four Advanced Space Engines (ASE) and requires a propellant capacity of 22.3 t. This concept can be adaptable to other DRMs — the lander can deliver 8.1 t to geostationary orbit (GEO) and the transfer vehicle can be used as a planetary propulsion unit. The total life cycle cost for this concept is $11.6 B. Similar summary sheets are shown in Figures 2.3.3.3-48 through 2.3.3.3-58 for the remaining five cargo and seven crew concepts.

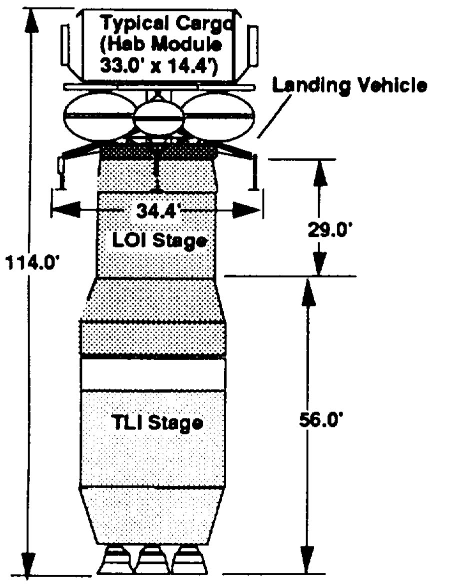


- Single Stage Transfer and Landing Vehicle
- Direct Earth to Lunar Surface Cargo Delivery
  No Return
  Lunar Architectures 2 & 3
- Transfer Vehicle -
  - 113.9 t Propellant
  - 4 RL-10 Engines
- Landing Vehicle
  - 22.3 t Propellant
  - 4 ASE's
- Launched Completely Assembled in a HLLV
  - Requires 36' Diameter Shroud
  - 147.4 t Performance Plus Cargo
- Evolution
  - Lander - Delivers 8.1 t to GEO
  - Transfer Vehicle - Planetary Propulsion Unit
- Program Cost
  - DDT&E - $6957M
  - Production - $1363M
  - Operations - $ 3304M
  - Total LCC - $11,624M
- No LEO Operations Required
- Cargo Height Above Lunar Surface - 20.4'
- Critical Operations - 1 Crit-1, 3 Crit-2

Preliminary Mass Properties

	(t)
Transfer Vehicle	6.1
Landing Vehicle	5.0
Total Mission Propellant	136.2

Figure 2.3.3.3-47 Configuration Summary Sheet - Concept 1A-1

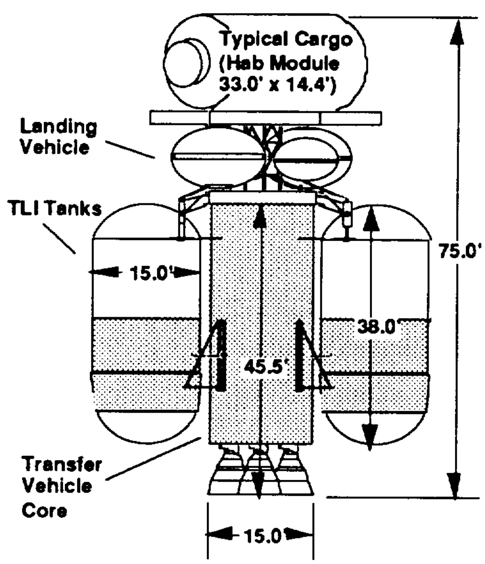


Preliminary Mass Properties

	(t)
TLI Stage	6.0
LOI Stage	3.0
Landing Vehicle	5.0
Total Mission Propellant	136.3

- Multi Stage Transfer Vehicle and Separate Landing Vehicle
- Direct Earth to Lunar Surface Cargo Delivery
  No Return
  Lunar Architectures 2 & 3
- TLI Stage
  - 95.3 t Propellant
  - 4 RL-10 Engines
- LOI Stage
  - 18.5 t Propellant
  - 2 RL-10 Engines
- Landing Vehicle
  - 22.5 t Propellant
  - 4 ASE's
- Launched Completely Assembled in a HLLV
  - Requires 36' Diameter Shroud
  - 150 t Performance Plus Cargo
- Evolution
  - Lander - Delivers 8.1 t to GEO
  - LOI Stage - Delivers 7.4 t to GEO
  - TLI Stage - Planetary Propulsion Unit
- Program Cost
  - DDT&E - $7234M
  - Production - $1533M
  - Operations - $ 3632M
  - Total LCC - $12,399M
- No LEO Operations Required
- Cargo Height Above Lunar Surface - 20.4'
- Critical Operations - 1 Crit-1, 4 Crit-2

Figure 2.3.3.3-48 Configuration Summary Sheet - Concept 1A-3



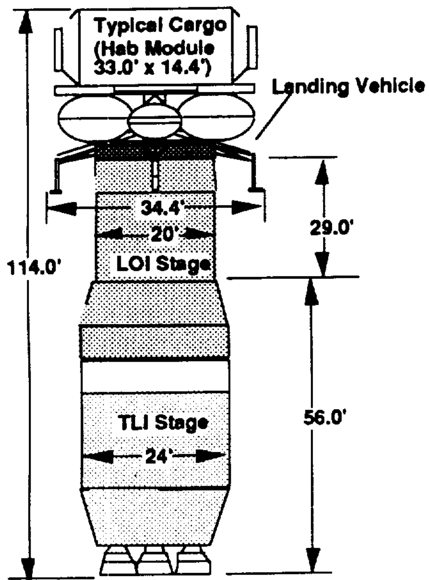
Preliminary Mass Properties

	(t)
Transfer Vehicle Core	4.0
TLI Tank (2 @ 2.5)	5.0
Landing Vehicle	5.0
Total Mission Propellant	133.5

Key Features

- Single Stage Transfer Vehicle with Drop Tanks
  Separate Landing Vehicle
- LEO to Lunar Surface Cargo Delivery
  No Return
  Lunar Architecture 1
- Transfer Vehicle Core -
  - 18.4 t Propellant
  - 4 RL-10 Engines
- Drop Tanks (2) TLI
  - 46.4 t Propellant (each)
- Landing Vehicle
  - 22.3 t Propellant
  - 4 ASE's
- Requires 2 Sh-C and 1 Sh-C Block 2 Flts for LEO Delivery
  - Transfer Core & Lander Packaged in Sh-C Block 2
  - Each TLI Tank Packaged in Sh-C
- Evolution
  - Lander - Delivers 8.1 t to GEO
  - Transfer Vehicle Core - Delivers 7.4 t to GEO
  - Transfer Core with Drop Tanks - Planetary Propulsion Unit
- Program Cost
  - DDT&E - $7657M
  - Production - $1560M
  - Operations - $3443M
  - Total LCC - $12,660M
- LEO Operations Include Delivery, Assy & Verification
  of Lander & Transfer Core & Drop Tanks
- Cargo Height Above Lunar Surface - 20.4'
- Critical Operations - 1 Crit-1, 4 Crit-2

Figure 2.3.3.3-49 Configuration Summary Sheet - Concept 3A-2



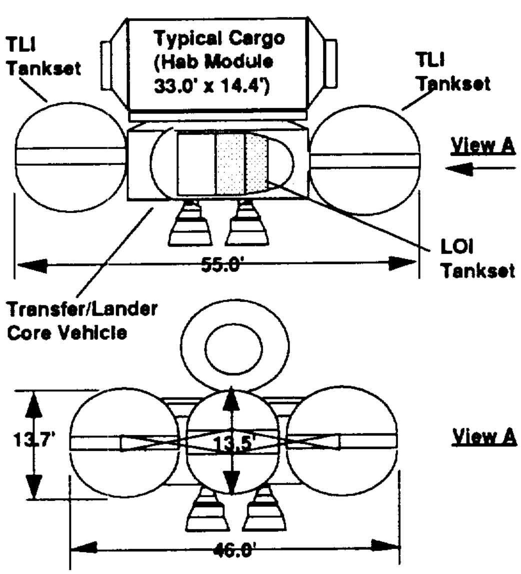
Preliminary Mass Properties

	(t)
TLI Stage	6.0
LOI Stage	3.0
Landing Vehicle	5.0
Total Mission Propellant	136.3

Key Features

- Multi Stage Transfer Vehicle and Separate Landing Vehicle
- LEO to Lunar Surface Cargo Delivery
  No Return
  Lunar Architecture 1
- TLI Stage
  — 95.3 t Propellant
  — 4 RL-10 Engines
- LOI Stage
  — 18.5 t Propellant
  — 2 RL-10 Engines
- Landing Vehicle
  — 22.5 t Propellant
  — 4 ASE's
- Requires 1 Sh-C Block 2 and 1 HLLV Fits for LEO Delivery
  — Lander and LOI Stage Packaged in Sh-C Block 2
  — TLI Stage Packaged in HLLV - 26' Dia., 101 t Perf
- Evolution
  — Lander - Delivers 8.1 t to GEO
  — LOI Stage - Delivers 7.4 t to GEO
  — TLI Stage - Planetary Propulsion Unit
- Program Cost
  — DDT&E - $10,551M
  — Production - $2011M
  — Operations - $3560M
  — Total LCC - $16,122M
- LEO Operations Include Delivery, Assy & Verification of Lander, LOI Stage and TLI Stage
- Cargo Height Above Lunar Surface - 20.4'
- Critical Operations - 1 Crit-1, 4 Crit-2

Figure 2.3.3.3-50 Configuration Summary Sheet - Concept 3A-3



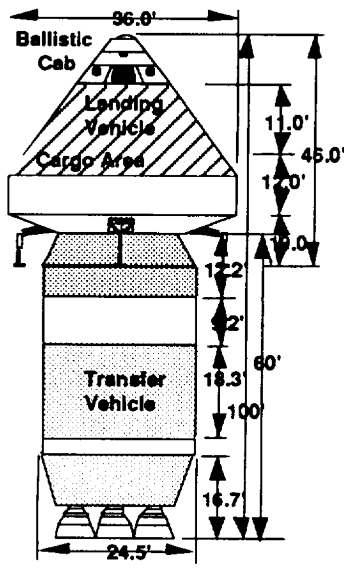
Preliminary Mass Properties

	(t)
Transfer/Vehicle Core	6.1
TLI Tank (2 @ 2.4)	4.8
LOI Tank (2 @ 1.2)	2.4
Total Mission Propellant	134.(

Key Features

- Single Propulsion Transfer/Landing Vehicle with Drop Tanks
- LEO to Lunar Surface Cargo Delivery
  No Return
  Lunar Architecture 1
- Transfer/Landing Vehicle Core -
  — 22.3 t Propellant
  — 4 RL-10 Engines
- Drop Tanks
  — (2) TLI 46.9 t Propellant (each)
  — (2) LOI 9 t Propellant (each)
- Requires 2 Sh-C and 1 Sh-C Block 2 Fits for LEO Delivery
  — Transfer/Landing Vehicle & LOI Tanks Packaged in Sh-C Block 2
  — Each TLI Tank Packaged in Sh-C
- Evolution
  — Transfer/Lander - Delivers 8.1 t to GEO
  — Transfer/Lander with Drop Tanks - Planetary Propulsion Unit
- Program Cost
  — DDT&E - $5997M
  — Production - $1255M
  — Operations - $ 3443M
  — Total LCC - $10,694M
- LEO Operations Include Delivery, Assy & Verification of Transfer/Lander Core and Drop Tanks
- Cargo Height Above Lunar Surface - 26.0'
- Critical Operations - 2 Crit-1, 3 Crit-2

Figure 2.3.3.3-51 Configuration Summary Sheet - Concept 3A-5



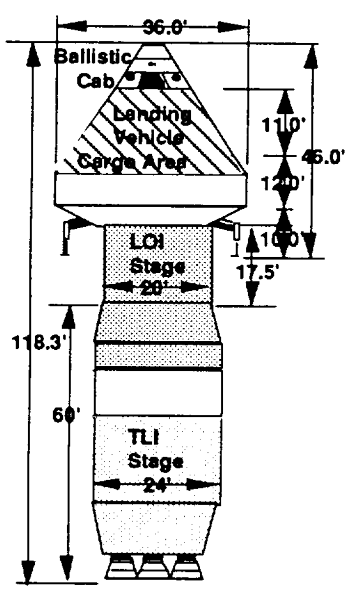
Preliminary Mass Properties

	(t)
Transfer Vehicle	6.7
Landing Vehicle	13.[illegible]
Total Mission Propellant	167.[illegible]

Key Features

- Single Stage Transfer Vehicle and Landing Vehicle
- Direct Earth to Lunar Surface Crew/Cargo Delivery
  Ballistic Return to Earth
  Lunar Architecture 3
- Transfer Vehicle -
  - 138 t Propellant
  - 4 RL-10 Engines
- Landing Vehicle
  - 29.2 t Propellant
  - 4 ASE's
- Launched Completely Assembled in a HLLV
  - Requires 36' Diameter Shroud
  - 188.8 t Performance Plus Cargo
- Evolution
  - Lander - Delivers 11.8 t to GEO
  - Transfer Vehicle - Planetary Propulsion Unit
- Program Cost
  - DDT&E - $8812M
  - Production - $7277M
  - Operations - $21,741M
  - Total LCC - $37,829M
- No LEO Operations Required
- Cargo Height Above Lunar Surface - 22'
- Critical Operations
  - Outbound - 1 Crit-1, 3 Crit-2
  - Return -5 Crit-1, 1 Crit-2

Figure 2.3.3.3-52 Configuration Summary Sheet - Concept 2D-1A



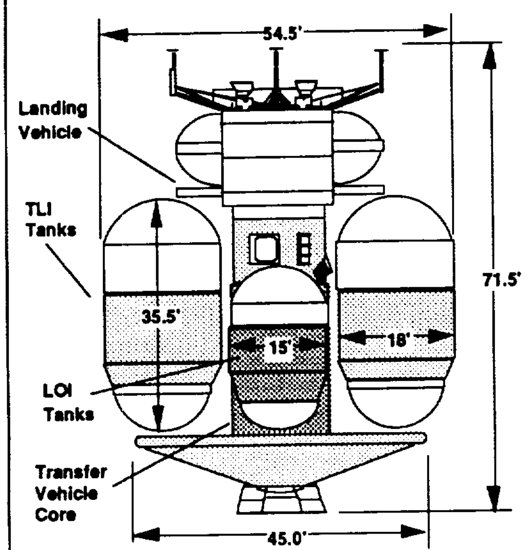
Preliminary Mass Properties

	(t)
TLI Stage	6.1
LOI Stage	3.3
Landing Vehicle	13.[illegible]
Total Mission Propellant	153.[illegible]

Key Features

- Multi Stage Transfer Vehicle and Landing Vehicle
- Direct Earth to Lunar Surface Crew/Cargo Delivery
  Ballistic Return to Earth / Lunar Architecture 3
- TLI Stage
  - 99.8 t Propellant
  - 4 RL-10 Engines
- LOI Stage
  - 24.6 t Propellant
  - 2 RL-10 Engines
- Landing Vehicle
  - 29.2 t Propellant
  - 4 ASE's
- Launched Completely Assembled in a HLLV
  - Requires 36' Diameter Shroud
  - 176.9 t Performance Plus Cargo
- Evolution
  - Lander - Delivers 11.8 t to GEO
  - LOI Stage - Delivers 9.4 t to GEO
  - TLI Stage - Planetary Propulsion Unit
- Program Cost
  - DDT&E - $9516M
  - Production - $8175M
  - Operations - $20,474M
  - Total LCC - $38,165M
- No LEO Operations Required
- Cargo Height Above Lunar Surface - 22'
- Critical Operations
  - Outbound - 1 Crit-1, 4 Crit-2
  - Return - 5 Crit-1, 1 Crit-2

Figure 2.3.3.3-53 Configuration Summary Sheet - Concept 2D-3A



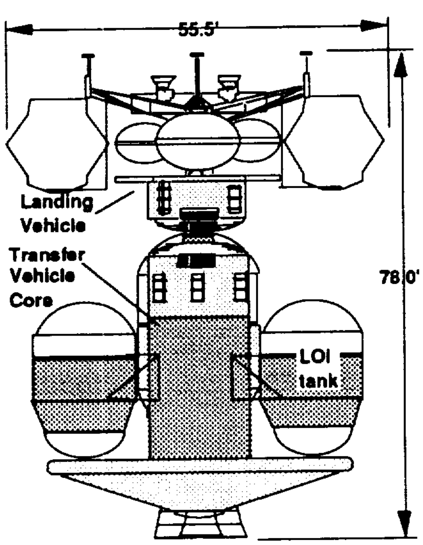
Key Features

- Single Stage Transfer Vehicle with Drop Tanks
  Separate Landing Vehicle
- LEO to Lunar Surface Crew/Cargo Delivery
  Aerobrake Return to LEO, Single Crew Cab
  Lunar Architectures 1 & 2
- Transfer Vehicle Core-5.8 t Propellant,4 RL-10 Engines
- Drop Tanks
  — (2) TLI 57.5 t Propellant (each)
  — (2) LOI 25.2 t Propellant (each)
- Landing Vehicle - 29.6 t Propellant, 4 ASE's
- Requires 2 Sh-C Block 2 & 2 HLLV Flts for LEO Deliv.
  — Transfer Core & Aerobrake Packaged in Sh-C Blk 2
  — Lander Packaged in Sh-C Block 2
  — TLI & LOI Tanks Pkgd - HLLV-20'Dia, 85t(Resupply)
- Evolution
  — Lander - Delivers 11.8 t to GEO
  — Transfer Core w/ Drop Tanks-Planetary Propul. Unit
- Program Cost
  — DDT&E - $10,060M
  — Production - $4296M
  — Operations - $20,154M
  — Total LCC - $34,510M
- LEO Ops Incl: Deliv., Assy, & Verif. of Lander, Transfer Core & Drop Tanks; Refurb Transfer Core & Crew Cab
- Cargo Height Above Lunar Surface - 6.5'
- Critical Operations
  — Initial Outbound - 1 Crit-1, 6 Crit-2
  — Steady State Outbound - 1 Crit-1, 8 Crit-2
  — Return - 6 Crit-1, 1 Crit-2

Preliminary Mass Properties

	(t)
Transfer Vehicle Core	13.(
TLI Tank (2 @ 2.7)	5.4
LOI Tank (2 @ 1.8)	3.6
Landing Vehicle	5.7
Total Mission Propellant	165.

Figure 2.3.3.3-54 Configuration Summary Sheet - Concept 4E-2A



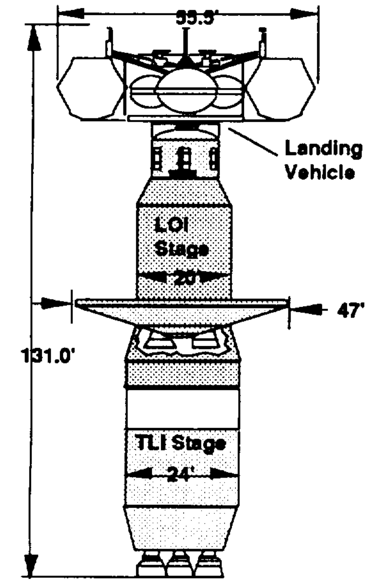
Key Features

- Single Stage Transfer Vehicle with Drop Tanks
  Separate Landing Vehicle
- LEO to Lunar Surface Crew/Cargo Delivery
  Aerobrake Return to LEO, Dual Crew Cabs
  Lunar Architectures 1 & 2
- Transfer Vehicle Core - 5.7 t Propellant, 4 RL-10 Engines
- Drop Tanks
  — (2) TLI 53.6 t Propellant (each)
  — (2) LOI 20.9 t Propellant (each)
- Landing Vehicle - 22.3 t Propellant, 4 ASE's
- Requires 2 Sh-C Block 2 and 2 HLLV Flights for LEO Delivery
  — Transfer Core & Aerobrake Packaged in Sh-C Block 2
  — Lander Packaged in Sh-C Block 2
  — Each TLI & LOI Tank Pkgd in HLLV - 20' Dia., 75 t (Resupply)
- Evolution
  — Lander - Delivers 8.1 t to GEO
  — Transfer Core with Drop Tanks - Planetary Propulsion Unit
- Program Cost
  — DDT&E - $10,431M
  — Production - $4699M
  — Operations - $18,402M
  — Total LCC - $33,532M
- LEO Operations Include Delivery, Assy, & Verification of Lander, Transfer Core & Drop Tanks; Refurb Transfer Core & Crew Cab
- Cargo Height Above Lunar Surface - 7.8'
- Critical Operations
  — Initial Outbound - 1 Crit-1, 7 Crit-2
  — Steady State Outbound - 1 Crit-1, 8 Crit-2
  — Return - 7 Crit-1, 1 Crit-2

Preliminary Mass Properties

	(t)
Transfer Vehicle Core	13.(
TLI Tank (2 @ 2.6)	5.2
LOI Tank (2 @ 1.6)	3.2
Landing Vehicle	8.6
Total Mission Propellant	148.

Figure 2.3.3.3-55 Configuration Summary Sheet - Concept 4E-2B



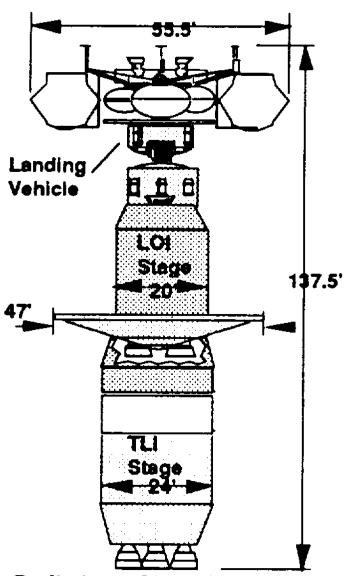
Key Features

- Multi Stage Transfer Vehicle, Separate Landing Vehicle
- LEO to Lunar Surface Crew/Cargo Delivery
  Aerobrake Return to LEO, Single Crew Cab
  Lunar Architectures 1 & 2
- TLI Stage 100 t Propellant, 4 RL-10 Engines
- LOI Stage - 53.5 t Propellant, 2 RL-10 Engines
- Landing Vehicle - 29.6 t Propellant, 4 ASE's
- Requires 2 Sh-C Block 2 and 1 HLLV Flts for LEO Delivery
  - — Transfer Core & Aerobrake Packaged in Sh-C Block 2
  - — Lander Packaged in Sh-C Block 2
  - — TLI Stage Packaged in HLLV - 25' Dia., 106 t
  - — Resupply Propellant for LOI Stage Packaged in Sh-C
- Evolution
  - — Lander - Delivers 12 t to GEO
  - — LOI Stage - Delivers 28.2 t to GEO, Delivery/Return 19.3 t to GEO
  - — TLI Stage - Planetary Propulsion Unit
- Program Cost
  - — DDT&E - $12,547M
  - — Production - $5198M
  - — Operations - $18,942M
  - — Total LCC - $36,687M
- LEO Ops Include Delivery, Assy, & Verification of Lander, TLI and LOI Stage; Refurb of LOI Stage & Crew Cab
- Cargo Height Above Lunar Surface - 6.5'
- Critical Operations
  - — Initial Outbound - 2 Crit-1, 3 Crit-2
  - — Steady State Outbound - 2 Crit-1, 6 Crit-2
  - — Return - 6 Crit-1, 1 Crit-2

Preliminary Mass Properties

	(t)
TLI Stage	6.1
LOI Stage	11.[illegible]
Landing Vehicle	5.7
Total Mission Propellant	153.[illegible]

Figure 2.3.3.3-56 Configuration Summary Sheet - Concept 4E-3A



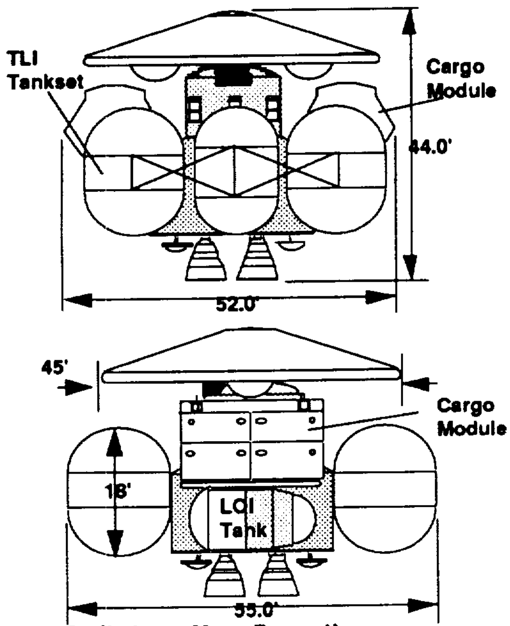
Key Features

- Multi Stage Transfer Vehicle, Separate Landing Vehicle
- LEO to Lunar Surface Crew/Cargo Delivery
  Aerobrake Return to LEO, Dual Crew Cabs
  Lunar Architectures 1 & 2
- TLI Stage 94.5 t Propellant, 4 RL-10 Engines
- LOI Stage - 45.5 t Propellant, 2 RL-10 Engines
- Landing Vehicle - 22.3 t Propellant, 4 ASE's
- Requires 2 Sh-C Block 2 and 1 HLLV Flts for LEO Delivery
  - — Transfer Core & Aerobrake Packaged in Sh-C Block 2
  - — Lander Packaged in Sh-C Block 2
  - — TLI Stage Packaged in HLLV - 25' Dia., 100 t
  - — Resupply Propellant for LOI Stage Packaged in Sh-C
- Evolution
  - — Lander - Delivers 8.1 t to GEO
  - — LOI Stage - Delivers 23 t to GEO, Delivery/Return 13.5 t to GEO
  - — TLI Stage - Planetary Propulsion Unit
- Program Cost
  - — DDT&E - $12,957M
  - — Production - $5649M
  - — Operations - $17,434M
  - — Total LCC - $36,040M
- LEO Operations Include Delivery, Assy, & Verif. of Lander, TLI and LOI Stage; Refurb of LOI Stage & Crew Cab
- Cargo Height Above Lunar Surface - 7.8'
- Critical Operations
  - — Initial Outbound - 2 Crit-1, 4 Crit-2
  - — Steady State Outbound - 2 Crit-1, 7 Crit-2
  - — Return - 7 Crit-1, 1 Crit-2

Preliminary Mass Properties

	(t)
TLI Stage	5.9
LOI Stage	12.[illegible]
Landing Vehicle	8.6
Total Mission Propellant	140.[illegible]

Figure 2.3.3.3-57 Configuration Summary Sheet - Concept 4E-3B



Key Features

- Single Propulsion Transfer/Landing Vehicle w/ Drop Tanks
- LEO to Lunar Surface Crew/CargoDelivery
  Aerobrake Return to LEO, Single Crew Cab
  Lunar Architectures 1 & 2
- Transfer/Landing Vehicle Core -
  - 29 t Propellant
  - 4 RL-10 Engines
- Drop Tanks
  - (2) TLI 66.5 t Propellant (each)
  - (2) LOI 10 t Propellant (each)
  - (2) Return Tankset 3 t Propellant (each)
- Requires 1 Sh-C Block 2 and 2 HLLV Flts for LEO Delivery
  - Transfer/Landing Vehicle & A/B Pkgd in Sh-C Block 2
  - Each TLI & Return Tankset Pkgd in HLLV - 20' Dia., 84 t
- Evolution
  - Transfer/Lander - Delivers 11.8 t to GEO
  - Transfer/Lander with Drop Tanks-Planetary Propul. Unit
- Program Cost
  - DDT&E - $10,130M
  - Production - $2866M
  - Operations - $19,153M
  - Total LCC - $32,149M
- LEO Operations Include Delivery, Assy & Verification of Core and Drop Tanks; Refurb of Core and Crew Cab
- Cargo Height Above Lunar Surface - 24.3'
- Critical Operations
  - Outbound - 1 Crit-1, 5 Crit-2
  - Return - 4 Crit-1, 1 Crit-2

Preliminary Mass Properties

	(t)
Transfer/Vehicle Core	16.
TLI Tank (2 @ 2.8)	5.
LOI Tank (2 @ 1.1)	2.
Total Mission Propellant	159.

Figure 2.3.3.3-58 Configuration Summary Sheet - Concept 4E-5B

2.3.3.3.2.6 Selection Criteria—Selection criteria and their associated weighting factors were developed prior to conducting the detailed evaluation of each configuration. Four selection criteria were utilized in the second downselect process—program cost, operational complexity, mission adaptability, and risk. These criteria are defined as listed below:

- Program Cost - the total cost to acquire and own the system including full scale development, verification, production, operations, support, performance, and disposal.
- Operational Complexity - addressed the number and complexity of the STV mission phases with the emphasis on safety and mission success.
- Mission Adaptability - determined the capability of a configuration to capture all or some of the STV design reference missions either with existing elements or the reconfiguration of an element.
- Risk - the probability of not meeting a technical, schedule, or cost requirement and the effect on the program if the requirement was not met.

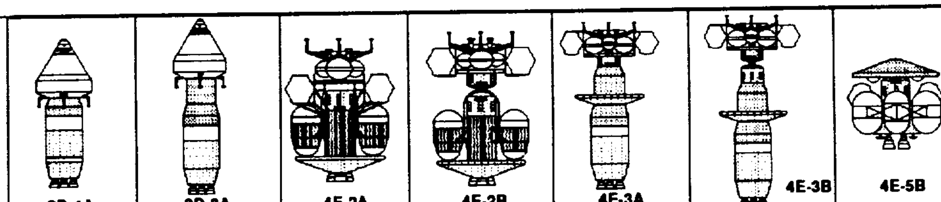
The data from the detailed concept definition was consolidated into four separate selection models—one for each criteria (one model emphasized cost as the primary driver, another emphasized operations, etc.). The evaluation values were then ranked in order of their value with the lowest value representing the best overall evaluation score. Selection of the final configurations were based on the best selection value from each criteria model.

The amount of influence that the results of the criteria/configuration evaluations had on the overall selection ranking of a configuration was determined by defining the weight that each criteria would carry during the selection analysis. These weight factors would be derived first as dictated by programs wants, and second by assigning a set value to a criteria and allowing the remaining criteria factors to shift according to program wants. A quality function deployment (QFD) analysis was used to develop both the derived set of weighting factors as well as the fixed values shown below:

Derived:	Cost = 50%, Ops = 30%, Mission Adapt = 2%, Risk = 18%
Fixed:	Ops = 50%, Cost = 25%, Mission Adapt = 5%, Risk = 20%
Fixed:	Risk = 50%, Cost = 20%, Ops = 25%, Mission Adapt = 5%
Fixed:	Mission Adapt = 50%, Cost = 15%, Ops = 20%, Risk = 15%

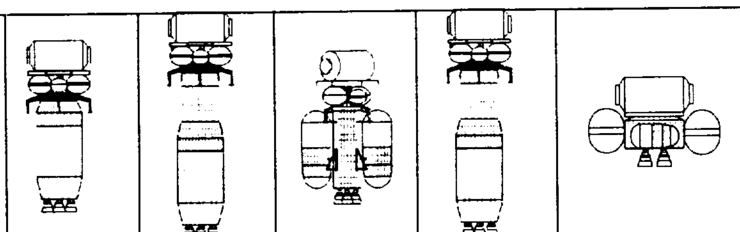
Following completion of this analysis, a review of the NASA criteria and their associated weighting factors showed a very close allocation.

2.3.3.3.2.7 Summaries/Recommendations—Figures 2.3.3.3-59 and 2.3.3.3-60 summarize the mass properties for the seven crew concepts and five cargo concepts evaluated during the second downselect process. A summary of the cost data for all twelve concepts is summarized in Figure 2.3.3.3-61. System elements costed for DDT&E, first unit, production, operations, and life cycle cost were vehicles—transfer (tanks, structure, propulsion, engines, and aerobrake) and lander (tanks, structure, propulsion, engines, and aerobrake), software, support equipment (GSE, ASE & SSE), tooling, system test and evaluation (ground, flight, operations), facilities, ground operations, flight operations, ETO costs, LEO node accommodations, and payloads. Ground processing operations analyses were based on the quantity of facility modifications and additions required to support the STV configuration as summarized in Figure 2.3.3.3-62. Processing timelines were determined not to be a discriminator at this analysis level. LEO node operations analyses as summarized in Figure 2.3.3.3-63 were based on the number of shifts and the complexity of the activities during the shift to process the STV. Analyses included vehicle refurbishment, crew module refurbishment, electrical checkout, engine servicing,



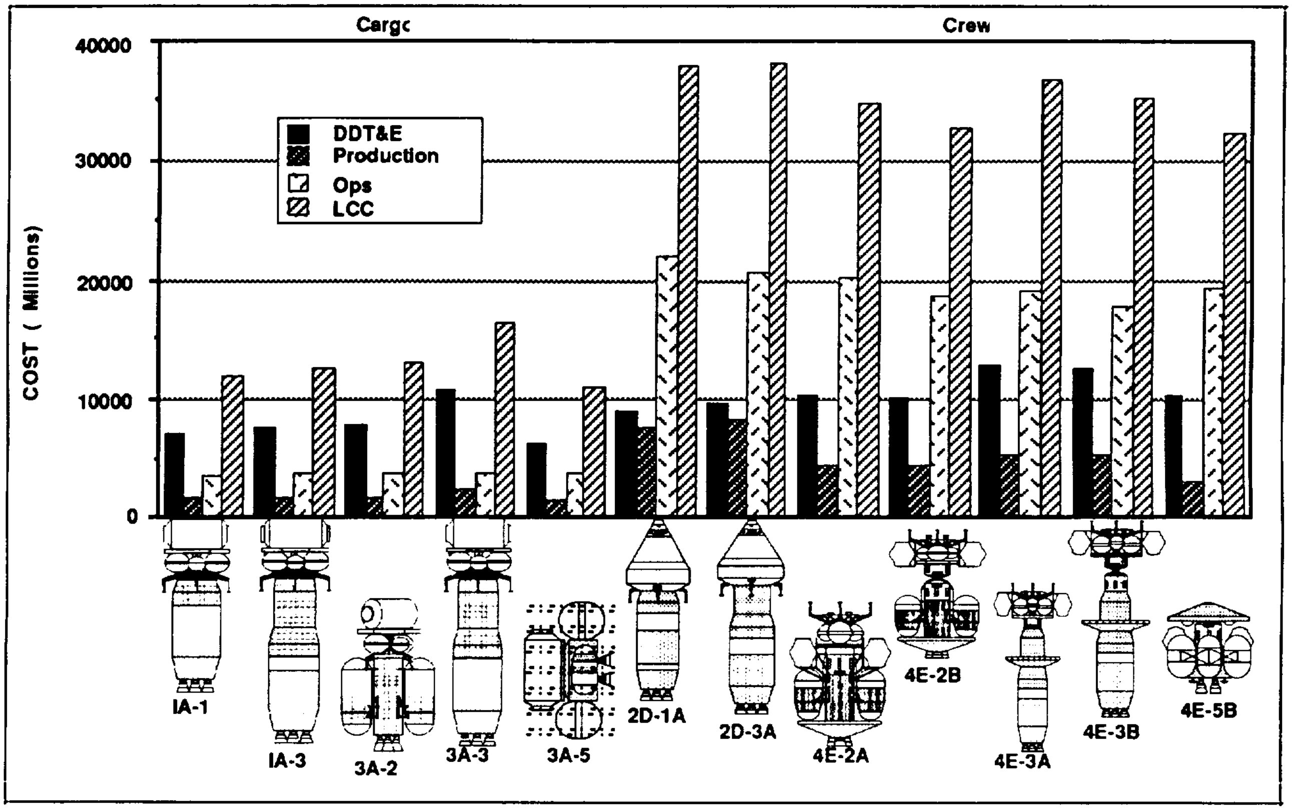
Crew Concepts w/ 14.6 t Cargo / Mass in t for Initial Flight	2D-1A	2D-3A	4E-2A	4E-2B	4E-3A	4E-3B	4E-5B
INERT MASS							
Transfer Vehicle Core/Stage 1	6.7	6.1	13.0	13.0	6.1	5.9	0.0
Transfer Vehicle Stage 2	0.0	3.3	0.0	0.0	11.8	12.1	0.0
TLI Drop Tanks	0.0	0.0	5.3	5.1	0.0	0.0	5.7
LOI Drop Tanks	0.0	0.0	3.6	3.3	0.0	0.0	2.2
Landing Vehicle	13.2	13.2	5.7	8.6	5.7	8.6	16.3
TOTAL INERT MASS	19.9	22.5	27.6	30.0	23.6	26.6	24.2
PROPELLANT MASS							
Transfer Vehicle Core/Stage 1	138.0	99.8	5.8	5.7	100.0	94.5	6.0
Transfer Vehicle Stage 2	0.0	24.6	0.0	0.0	23.9	23.2	0.0
TLI Drop Tanks	0.0	0.0	109.2	101.4	0.0	0.0	104.0
LOI Drop Tanks	0.0	0.0	20.8	19.5	0.0	0.0	20.0
Landing Vehicle	29.2	22.5	29.6	22.3	29.6	22.3	29.0
TOTAL PROPELLANT MASS	167.2	153.6	165.4	148.9	153.5	140.0	159.0
TOTAL VEHICLE MASS	187.1	176.1	193.0	178.9	177.1	166.6	183.2

Figures 2.3.3.3-59 Crew Concept - Mass Properties Summary

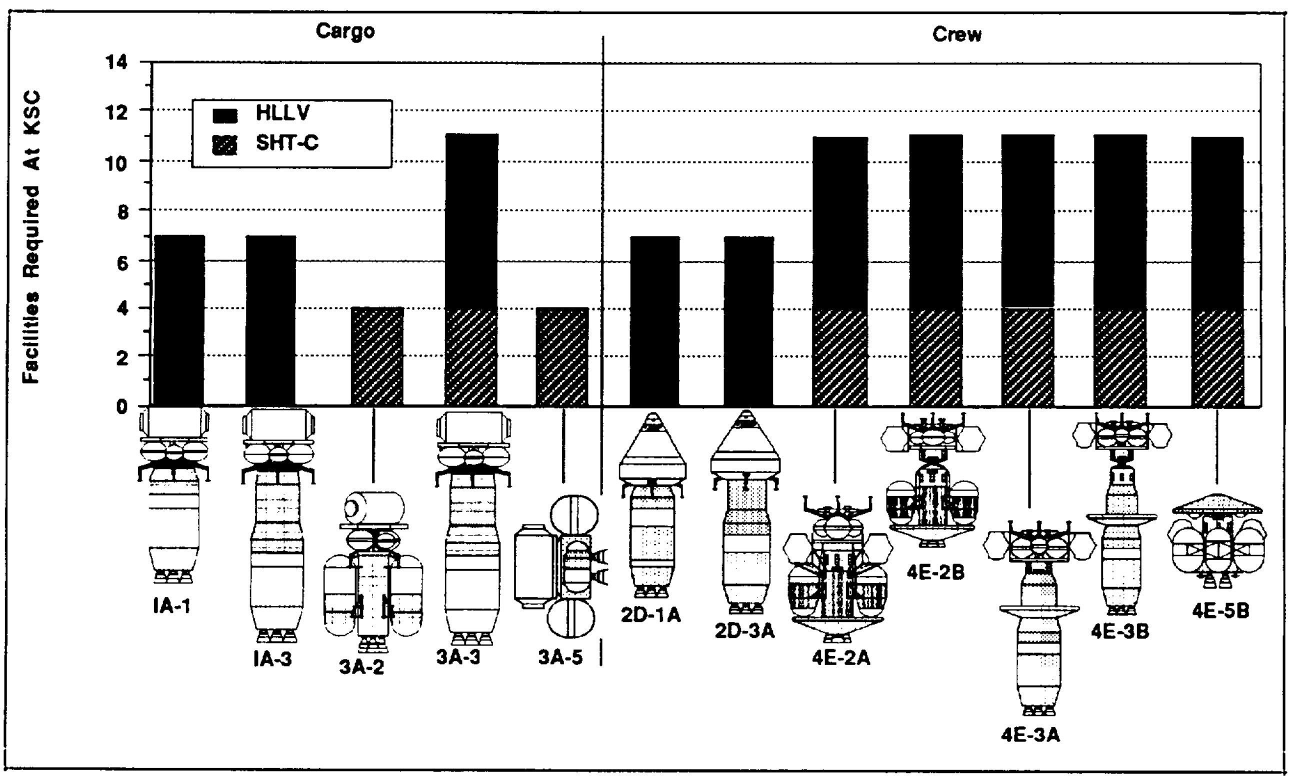


Cargo Concepts w/ 33 t Cargo / Mass in t for ALL FlightS	1A-1	1A-3	3A-2	3A-3	3A-5
INERT MASS					
Transfer Vehicle Core/Stage 1	6.1	6.0	4.0	6.0	0.0
Transfer Vehicle Stage 2	0.0	3.0	0.0	3.0	0.0
TLI Drop Tanks	0.0	0.0	5.0	0.0	4.8
LOI Drop Tanks	0.0	0.0	0.0	0.0	2.4
Landing Vehicle	5.0	5.0	5.0	5.0	6.1
TOTAL INERT MASS	11.1	14.1	14.0	14.1	13.3
PROPELLANT MASS					
Transfer Vehicle Core/Stage 1	113.9	95.3	18.4	95.3	0.0
Transfer Vehicle Stage 2	0.0	18.5	0.0	18.5	0.0
TLI Drop Tanks	0.0	0.0	92.8	0.0	93.8
LOI Drop Tanks	0.0	0.0	0.0	0.0	17.9
Landing Vehicle	22.3	22.5	22.3	22.3	22.3
TOTAL PROPELLANT MASS	136.2	136.3	133.5	136.3	134.0
TOTAL VEHICLE MASS	147.3	150.4	147.5	150.4	147.3

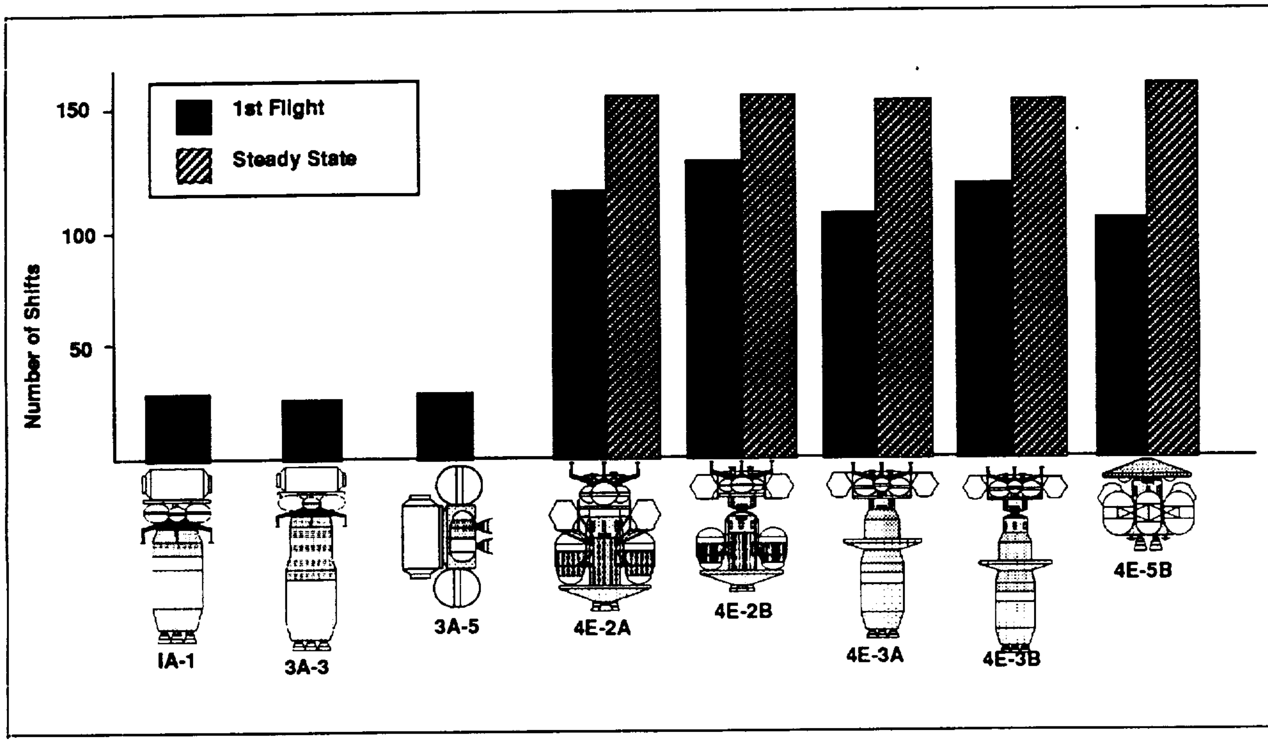
Figures 2.3.3.3-60 Cargo Concept - Mass Properties Summary



Figures 2.3.3.3-61 Cost Evaluation Summary



Figures 2.3.3.3-62 Ground Operations Evaluation Summary



Figures 2.3.3.3-63 LEO Node Operations Complexity Summary

aerobrake TPS servicing, subsystem leak and functional checks, TCS refurbishment, avionics system verification, hardware delivery, assembly, verification, propellant servicing, closeout, launch, and post flight. An evaluation of the critical space transfer operations was based on the quantity of criticality 1 and 2 mission operations for each configuration for both the first flight and steady state flights. Figure 2.3.3.3-64 presents the results of this evaluation which included rendezvous and docking, engine burns, separations, crew transfers, cargo transfers, propellant transfers, and other maneuvers such as closing aerobrake doors and aerobraking. A mission adaptability analysis was based on the assessment of the configuration to support the STV design reference missions and the capability of the configuration to implement changes in technology without an impact to the mission. Figure 2.3.3.3-65 presents the mission adaptability analysis summary. Figure 2.3.3.3-66 summarizes the results of the risk evaluation analysis which was based on a qualitative assessment of the probability of not meeting a technical, schedule, or cost requirement and the overall program effect of not meeting that requirement. Using the quantitative values produced from the criteria-based selection models, each of the configurations were ranked in order of lowest selection value to highest (lowest being the best). For the piloted configurations, this produced a ranking of from one to seven and in the cargo configurations, a ranking of one to five. This was done for each of the four selection criteria, producing the relative selection ranking chart illustrated in Figure 2.3.3.3-67.

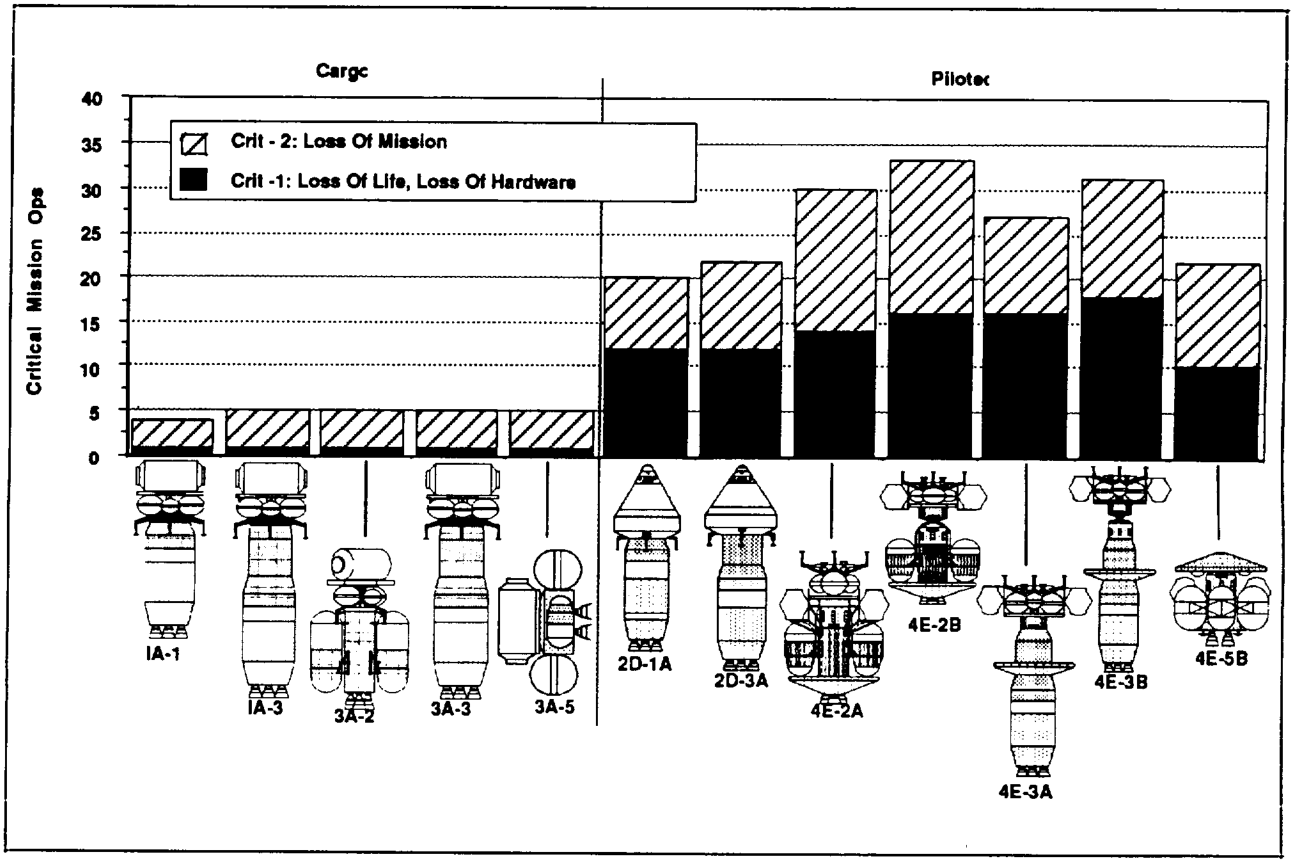


Figure 2.3.3.3-64 Space Transfer Critical Operations Summary

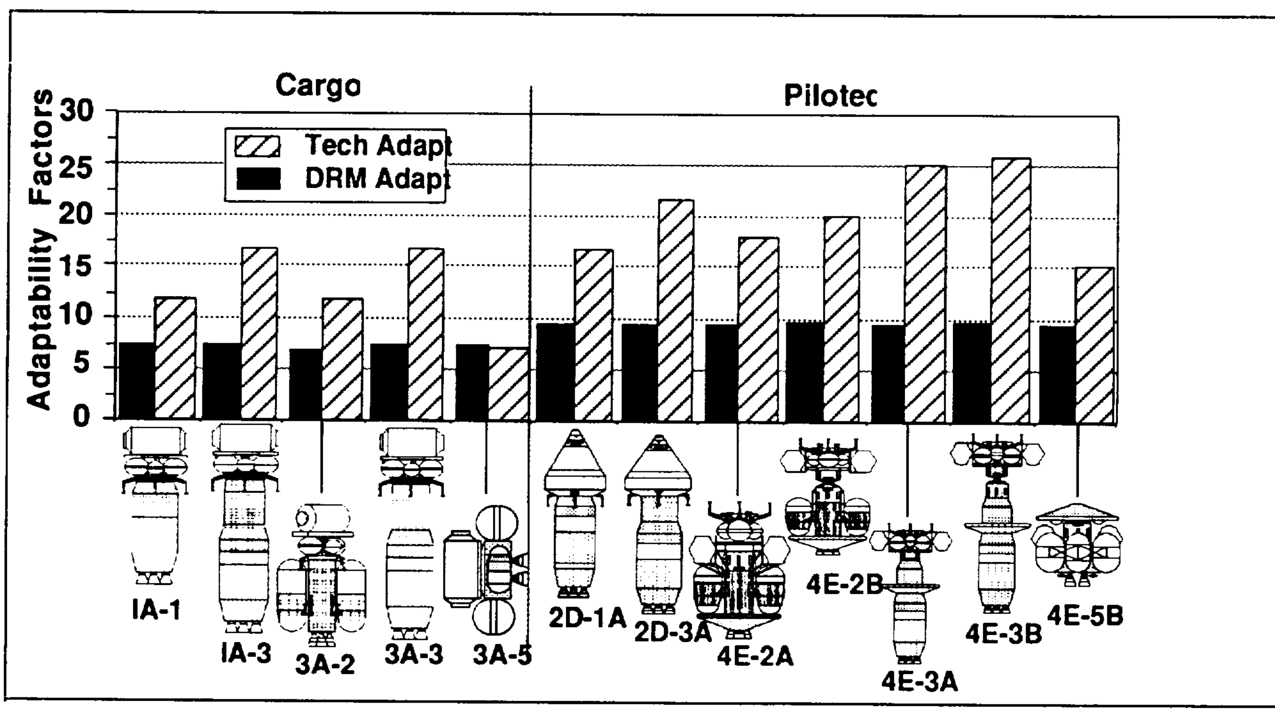


Figure 2.3.3.3-65 Mission Adaptability Evaluation Summary

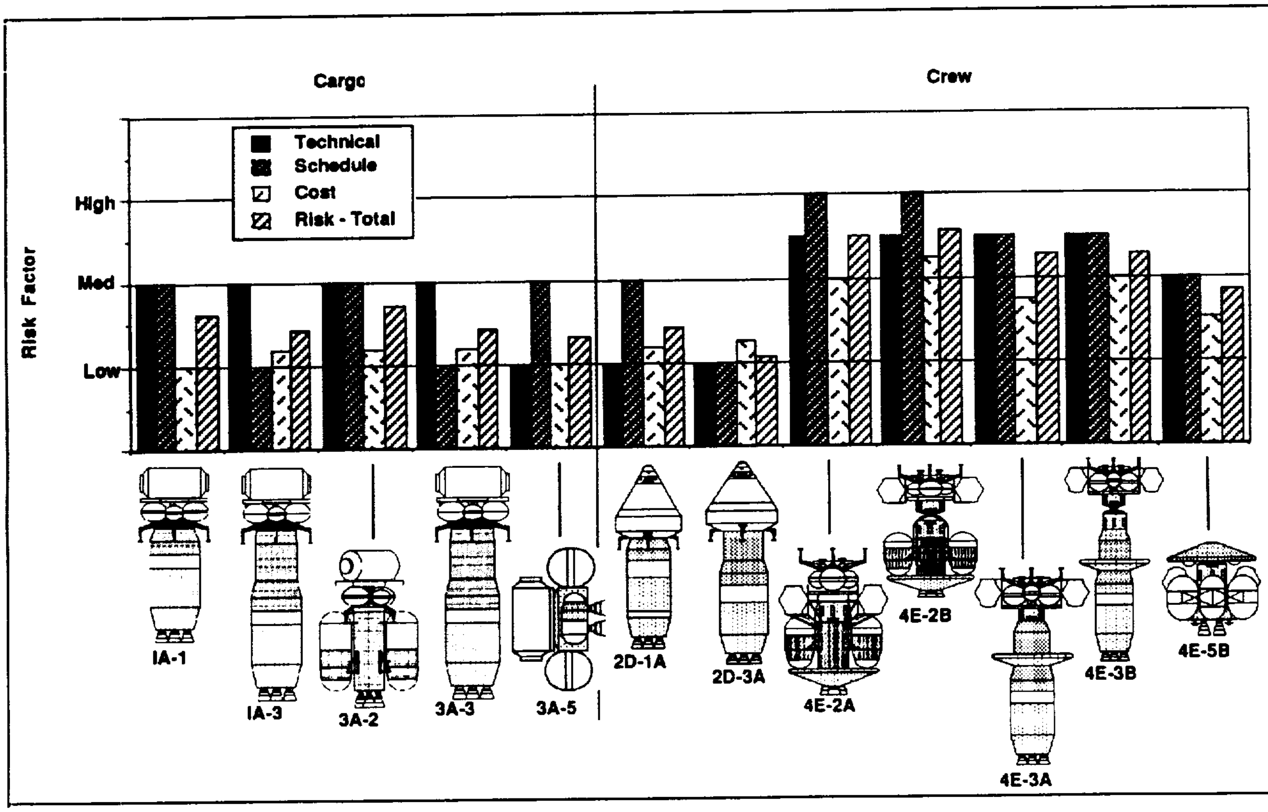


Figure 2.3.3.3-66 Risk Evaluation Summary

	Cargo Evaluation Rankir					Crew Evaluation Rankin						
Cost Driv	2	5	3	4	1	6	5	7	3	4	2	1
Operations Driven	2	5	3	4	1	6	4	7	3	5	2	1
Adaptability Driven	2	5	3	4	1	4	5	6	7	3	2	1
Risk Driven	2	5	3	4	1	6	5	7	3	4	2	1

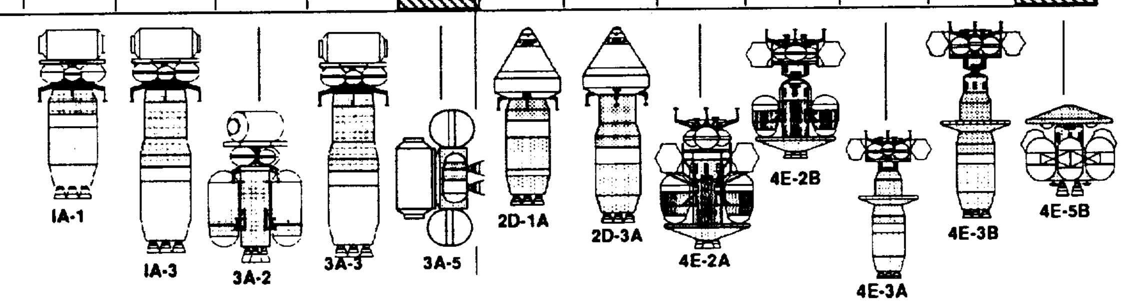


Figure 2.3.3.3-67 Configuration Selection Evaluation Summary

The single stage transfer and separate lander cargo concept (1A-1) was the second best cargo configuration against all selection criteria. The multi-stage transfer and separate lander crew concept (4E-3B) was the second best crew configuration with the 90-day reference configuration (4E-2B) as the third best piloted configuration.

Based on the results of the second downselect process, five vehicle configurations were recommended for additional study during the final downselect process. These five configurations (two cargo and three piloted/cargo configurations) are shown in Figure 2.3.3.3-68.

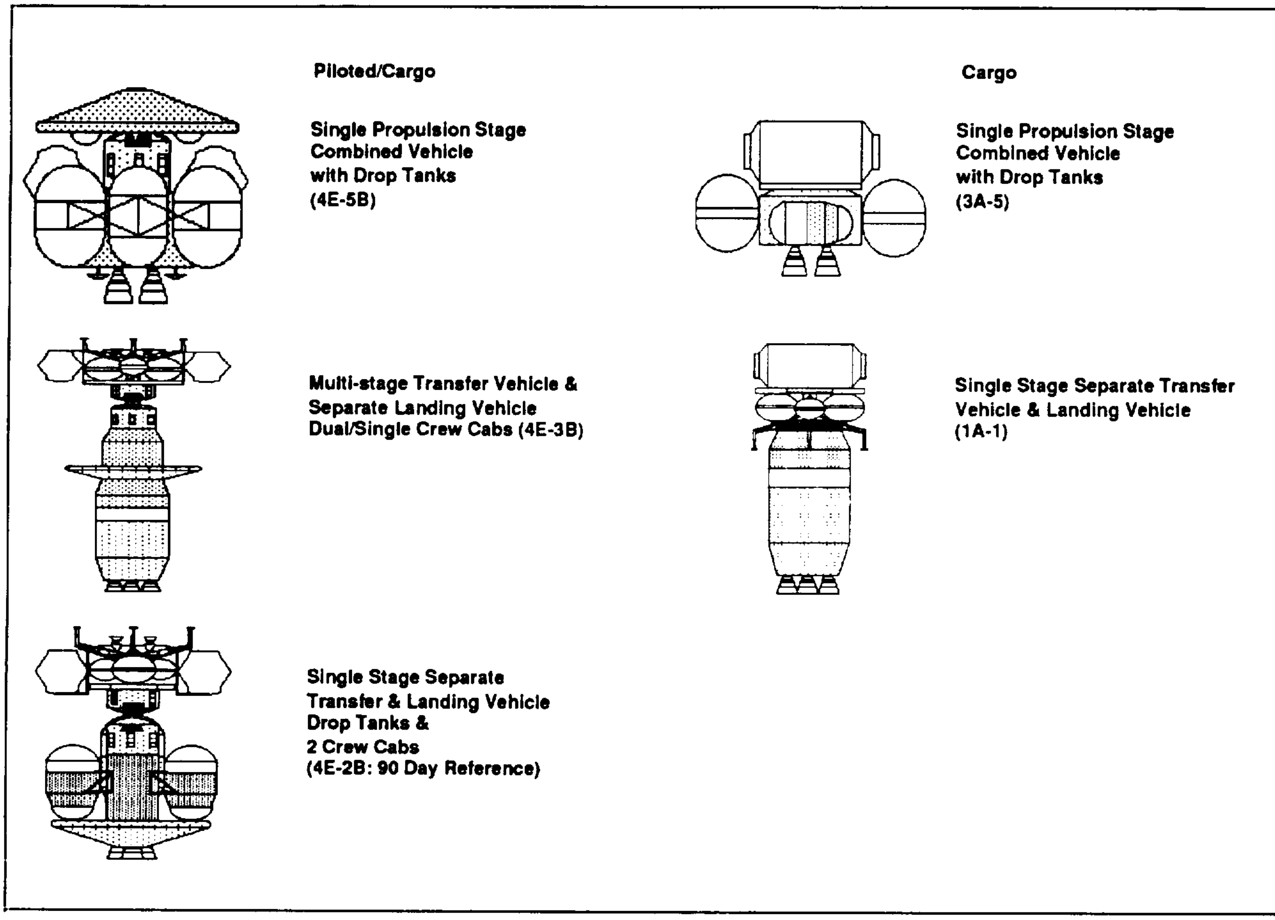


Figure 2.3.3.3-69 Recommended Configurations

2.3.3.3.3 Final Downselect Process—

2.3.3.3.3.1 Methodology—Figure 2.3.3.3-70 illustrates the overall methodology employed during the final downselect process. The final phase of the concept selection trade started with determining the feasibility of combining the piloted and cargo versions of the configurations recommended from the second downselect into common vehicles. Following the commonality evaluation, a final configuration analysis was performed to select the final recommended configuration. This final evaluation was based on an operational contingency analysis and a detailed cost/operations analysis. After the selection of the recommended STV for the lunar transportation mission, a configuration based reusability trade was conducted.

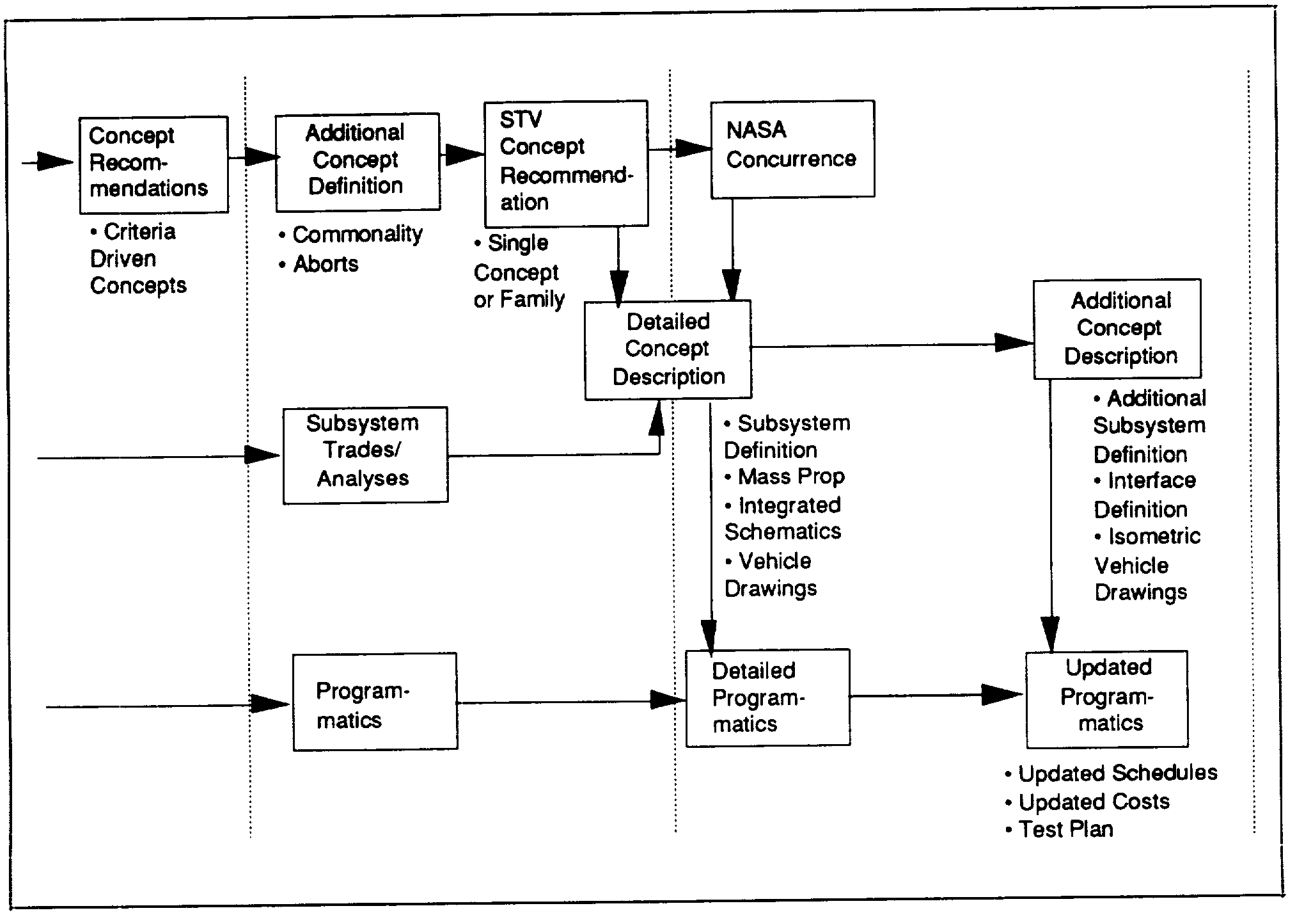


Figure 2.3.3.3-70 Methodology - Final Downselection

2.3.3.3.3.2 Groundrules & Assumptions—The following top level groundrules and assumptions were utilized during the final downselect process:

- Lunar DRMs were used to define the initial concepts. Downselected concepts were evaluated for adaptability to meet all DRMs.
- Cargo only expendable concepts were rated separately from crew and cargo missions. Commonality evaluation to determine common piloted and cargo vehicles.
- Cargo delivery requirements were 33 t and crew/cargo delivery requirements were 14.6 t.
- Subsystem definition was constant across all concepts, i.e., rigid aerobrake on all LEO return concepts, advanced space engines on landing vehicles, etc.

- Initial flight and steady state flights were evaluated for all concepts.
- No constraints were placed on the Earth-to-orbit (ETO) transportation system.
- All crew and cargo missions utilized LLO operations.
- All piloted missions use a LEO Assembly Node.

2.3.3.3.3.3 Commonality Analysis—Up until the final downselect process, cargo vehicles were evaluated separately from piloted vehicles. The first phase of the final downselect process was to determine the feasibility of combining the piloted and cargo versions of a configuration into common vehicle families. The five configurations (2 cargo and 3 piloted) recommended for additional study from the second downselect were evaluated to determine commonality between the vehicle elements. The thrust of this assessment was to breakdown each cargo and piloted configuration into similar components and evaluate the commonality between them. The elements that formulated the basis for the evaluation were:

- Tanks/Stages
    - TLI Drop Tanks
    - LOI Drop Tanks
    - TLI Stages
    - LOI Stages
- Vehicles
    - LTV
    - LEV
- Engines
    - Type
    - Quantity
    - Thrust Levels

The piloted and cargo concepts recommended from the second downselect were combined to form three common families of vehicles. The propellant quantities required to perform the piloted and cargo missions were determined. Common tankage, stages, and landers were sized to meet the maximum propellant requirements. Propellant will be offloaded when not required for the particular mission. The commonality assessment showed that each of the piloted configurations could support the expendable cargo missions. The single stage separate transfer and landing vehicle configuration has a propellant capacity of 148.9 t in the piloted mode. When the aerobrake and crew module are replaced with 33.0 t of cargo, the propellant requirements drop to 142 t. This trend held true for the remaining two candidates; the single propulsion stage required 158.6 t in the piloted mode and 147.2 t for cargo, the multi-stage transfer separate stage landing vehicle required

140 t in the piloted mode and 135.0 t for cargo. If the cargo delivery requirement of 33.0 t was used as the discriminator, each of the configurations would not meet the 14.6 t payload delivery plus crew delivery requirement. Figure 2.3.3.3-71 illustrates the three common families and their required propellant quantities.

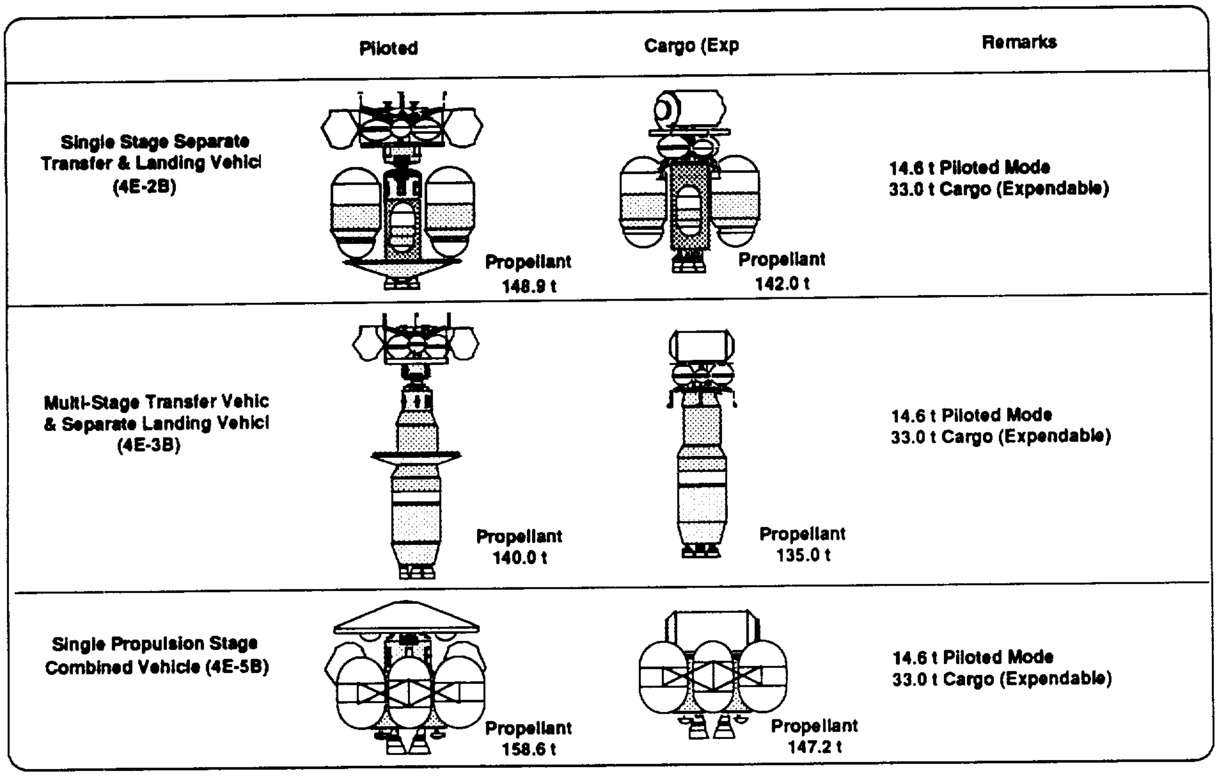


Figure 2.3.3.3-71 Common Families

2.3.3.3.3.4 Operational Contingency Analysis—The next phase of the final downselect process was to conduct an operational contingency analysis. This analysis addressed each lunar mission phase, determined possible contingencies for system failures, and provided a recommendation on which of the configurations tended to have the fewest mission anomalies. The mission consisted of an initial flight out-bound leg, an initial/steady state in-bound leg, and a steady state out-bound leg. The out-bound leg for an initial flight configuration initiates with the TLI preparation and burn, after which the TLI tanks are dropped and several mid course correction are made. This is followed by LOI, separation of the landing and LLO elements, and descent to the Lunar surface. The in-bound leg initiates with ascent from the lunar surface to LLO, where the lander rendezvous and docks with the LLO element. Following docking, the system performs TEI, conducts mid-course correction and performs reentry and rendezvous with the LEO node. The out-bound steady state mission duplicates the initial flight mission except that the transfer system must rendezvous and dock in LLO with the lander. Figure 2.3.3.3-72 presents the lunar mission operational scenario.

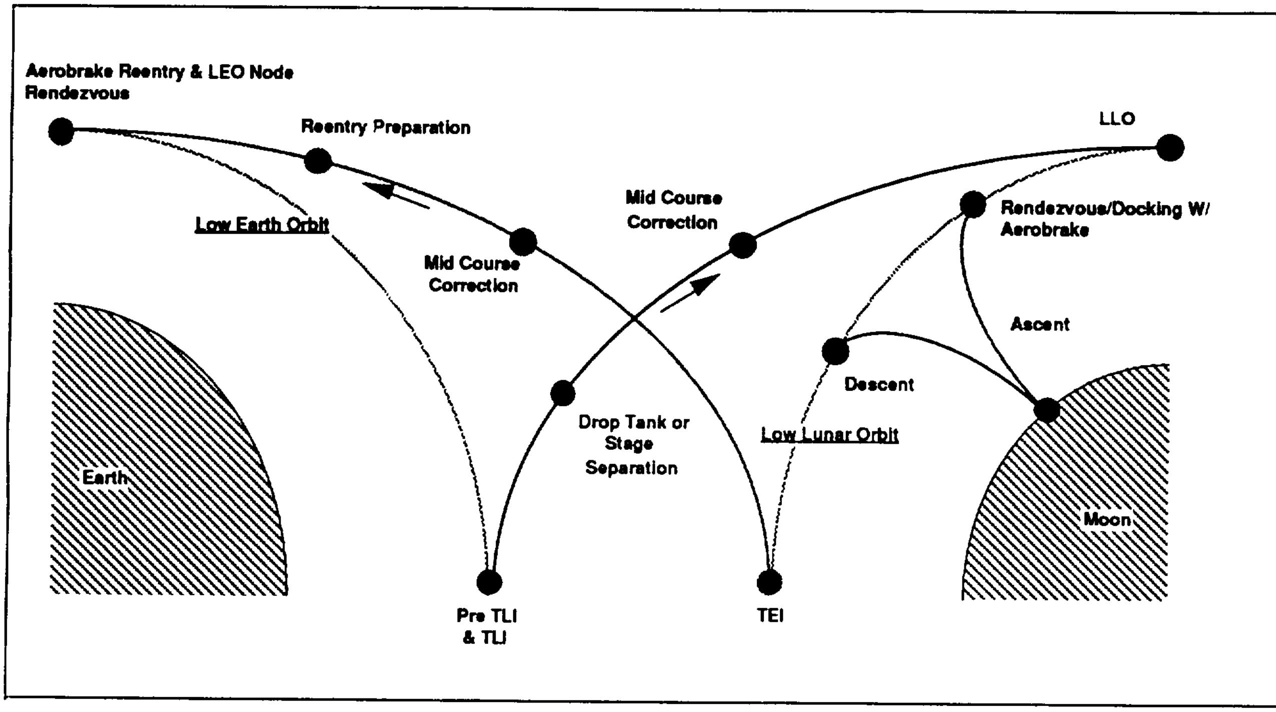


Figure 2.3.3.3-72 Lunar Mission Operational Scenario

The detailed analysis for the initial flight outbound mission illustrated in Figure 2.3.3.3-73 produced a single discriminator with the single propulsion system during LLO ops. Because of the single propulsion system, if a failure occurred in the propulsion system, there are no alternatives for completing the mission without the use of a rescue flight. For the remainder of the mission elements, each configuration provided a similar mission contingency. During the steady state outbound flight (Fig. 2.3.3.3-74), the single propulsion stage poses the same issues as found in the initial flight. The other configurations must rendezvous and dock with the lander in LLO. Failure to do this results in loss of mission.

<table>
<tr><th>Abort Initiated</th><th>Pre TLI</th><th>During TLI Burn</th><th>Separation of Drop Tanks or Stage</th><th>Mid-Course Correction Burn</th><th>Pre -LLO Insertion Burn</th><th>During LLO Insertion Burn</th><th>In LLO Prior to Landing</th><th>During Landing Burn</th></tr>
<tr><td>Single Vehicle Family (4E-2B)</td><td rowspan="3">Remain in Low Earth Orbit and rendezvous with orbiting platform</td><td rowspan="2">Use Landing vehicle or second stage of transfer vehicle to return to LEO</td><td rowspan="3">Use Landing Vehicle engines for propulsive reentry to LEO</td><td rowspan="3">Use TV or LV RCS or LV main propulsion systems</td><td rowspan="3">Free return trajectory normal aerobrake reentry</td><td rowspan="2">Use Landing Vehicle to achieve LLO or perform TEI burn</td><td rowspan="2">Use Landing Vehicle engines for TEI burn</td><td rowspan="3">Use second pair of engines and continue landing burn or abort to LLO</td></tr>
<tr><td>Multi-Stage Family (4E-3B)</td></tr>
<tr><td>Single Propulsion Family (4E-5B)</td><td>Use RCS as far as possible to return to LEO. Not possible thru complete burn</td><td>Use RCS to achieve LLO or perform TEI burn</td><td>No option for propulsion system loss. Rescue mission required.</td></tr>
</table>

Figure 2.3.3.3-73 Initial Flight Outbound Mission

<table>
<tr><th>Abort Initiated</th><th>Pre TLI</th><th>During TLI Burn</th><th>Separation of Drop Tanks or Stage</th><th>Mid- Course Correction Burn</th><th>Pre-LLO Insertion Burn</th><th>During LLO Insertion Burn</th><th>Rendez-vous & Docking</th><th>In LLO Prior to Landing</th><th>During Landing Burn</th></tr>
<tr><td>Single Vehicle Family (4E-2B)</td><td rowspan="3">Remain in Low Earth Orbit and rendez-vous with orbiting platform</td><td rowspan="3">Use RCS as far as possible to return to LEO or complete TLI burn. Not possible thru complete burn.</td><td>Fly free return trajectory and use Transfer Vehicle engines for propulsive reentry to LEO</td><td rowspan="3">Use Transfer Vehicle RCS</td><td rowspan="3">Free return trajectory normal aerobrake reentry</td><td rowspan="3">Use RCS to achieve LLO or perform TEI burn. Not possible thru complete burn</td><td rowspan="2">Perform TEI burn and return to LEO</td><td rowspan="2">Use Landing Vehicle engines for TEI burn</td><td rowspan="3">Use second set of engines and continue landing burn or abort to LLO</td></tr>
<tr><td>Multi-Stage Family (4E-3B)</td><td>EVA for possible manual separation. No option. In-flight rescue unlikely</td></tr>
<tr><td>Single Propulsion Family (4E-5B)</td><td>Use Landing Vehicle engines for propulsive reentry to LEO</td><td>N/A</td><td>No option for propulsion system loss. Rescue mission required</td></tr>
</table>

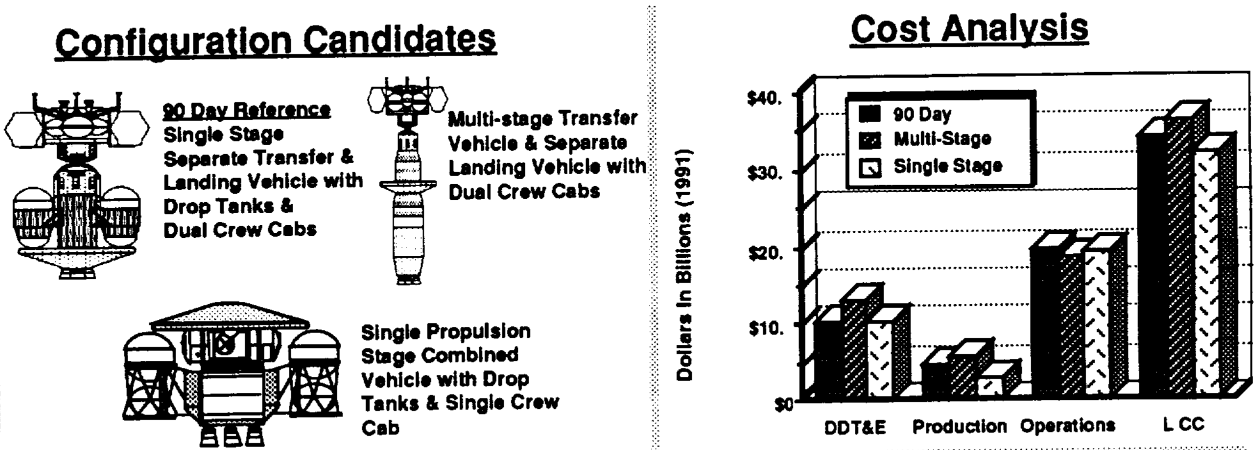
Figure 2.3.3.3-74 Steady State Outbound Mission

An assessment of the inbound mission (Figure 2.3.3.3-75 produced a single discriminator in the performance of each configuration. When the single propulsion system is rendezvousing and docking with the LLO element, if the aerobrake is damaged or lost, the lander is stranded in LLO and a rescue mission is required. This scenario is true for the other configurations, although it may be possible to conduct an EVA transfer of the crew to the transfer vehicle. When the single stage separate vehicle and multi-stage transfer and separate landing vehicle are aerobraking at earth, if the engine doors in the aerobrake cannot be closed, a rescue mission is required. Results of the contingency analysis showed no clear discriminators between the candidates. Since each of the configurations has advantages and disadvantages, there was no configuration that stood out as being better than the others.

<table>
<tr><th>Abort Initiated</th><th>Prior to Lunar Lift-Off</th><th>During Ascent Burn</th><th>Rendezvous and Docking</th><th>In LLO</th><th>During TEI Burn</th><th>Mid-Course Correction Burn</th><th>Close Aerobrake Doors</th><th>Aero-brake Reentry</th><th>Prior to Docking with LEO Node</th></tr>
<tr><td>Single Vehicle Family (4E-2B)</td><td rowspan="3">Vehicle and crew remain on lunar surface. Crew reenters Habitat Module and awaits rescue mission</td><td rowspan="3">Use second set of engines and continue ascent burn or return to Lunar surface and await rescue mission</td><td rowspan="2">Use EVA to transfer crew to TV and continue mission</td><td rowspan="3">Return to lunar surface or use LV engines for TEI burn</td><td rowspan="3">Use RCS as far as possible to return to LLO or obtain LEO. Not possible thru complete burn</td><td rowspan="3">Use RCS</td><td rowspan="2">EVA to attempt manual closure. No option unless residual fuel sufficient for propulsive reentry</td><td rowspan="3">Propulsive reentry if sufficient propellants remain</td><td rowspan="3">Utilize RCS to rendezvous with station or rescue mission is required</td></tr>
<tr><td>Multi-Stage Family (4E-3B)</td></tr>
<tr><td>Single Propulsion Family (4E-5B)</td><td>No option. LLO rescue mission required</td><td>N/A</td></tr>
</table>

Figure 2.3.3.3-75 Inbound Mission

2.3.3.3.3.5 Cost/Operations Analysis—The last phase of the final downselect process was to perform a detailed analysis of system costs and operations. The cost evaluation was based on DDT&E, production, operations, and life cycle costs. As shown in Figure 2.3.3.3-76, the single propulsion family (4E-5B) had the lowest life cycle costs. The operations analysis addressed ground facility impacts, LEO assembly and checkout, and mission failure modes. The single propulsion family also exhibited the lowest number of shifts required for initial flight assembly and checkout at Space Station Freedom. When analyzing the cost and operations data for each configuration, the weighting factors that were developed during the preliminary configuration analysis were incorporated. Based on the weighted values determined during the study, the single propulsion system family was the clear winner.



Operations Analysis

		90-Day	Single Stage	Multi-Stage
Ground	Processing Facilities Mods And/Or Additions	9	9	9
LEO	Initial Flight A&CO Complexity Steady State Flight A&CO Complexity	125 Shifts 150 Shifts	100 Shifts 160 Shifts	115 Shifts 150 Shifts
Mission Failure Modes	Crit 1 Crit 2	15 16	10 12	17 12

Figure 2.3.3.3-76 Life Cycle Cost/Operations Data

2.3.3.3.3.6 Reusability Analysis—After the final configuration selection was complete, a configuration reusability trade was conducted to address the feasibility of reusing the vehicles for cargo missions (Fig. 2.3.3.3-77). Performance data defined a cargo capacity range of from 37.4 t for expendable missions, to 25.9 t for a reusable cargo mission, to 14.6 t for a piloted mission. Because the 25.9 t does not comply with the 33.0 t cargo requirement, an evaluation of the actual payload support systems manifested cargo indicated that the 25.9 t capability is within the noise range of the actual mass requirements of 26.46 t. Based on this, the recommendation to reuse the cargo vehicles based on performance is a valid one.

The final piece of data that was required to complete the reusability study was the economic impact of reusing the cargo vehicle (Fig. 2.3.3.3-78). With the reuse of one of the four cargo only vehicles that are currently manifested in Option 5, the total lunar transfer system vehicle requirement is reduced from nine to eight. The cost saving associated with this reduction is $0.8

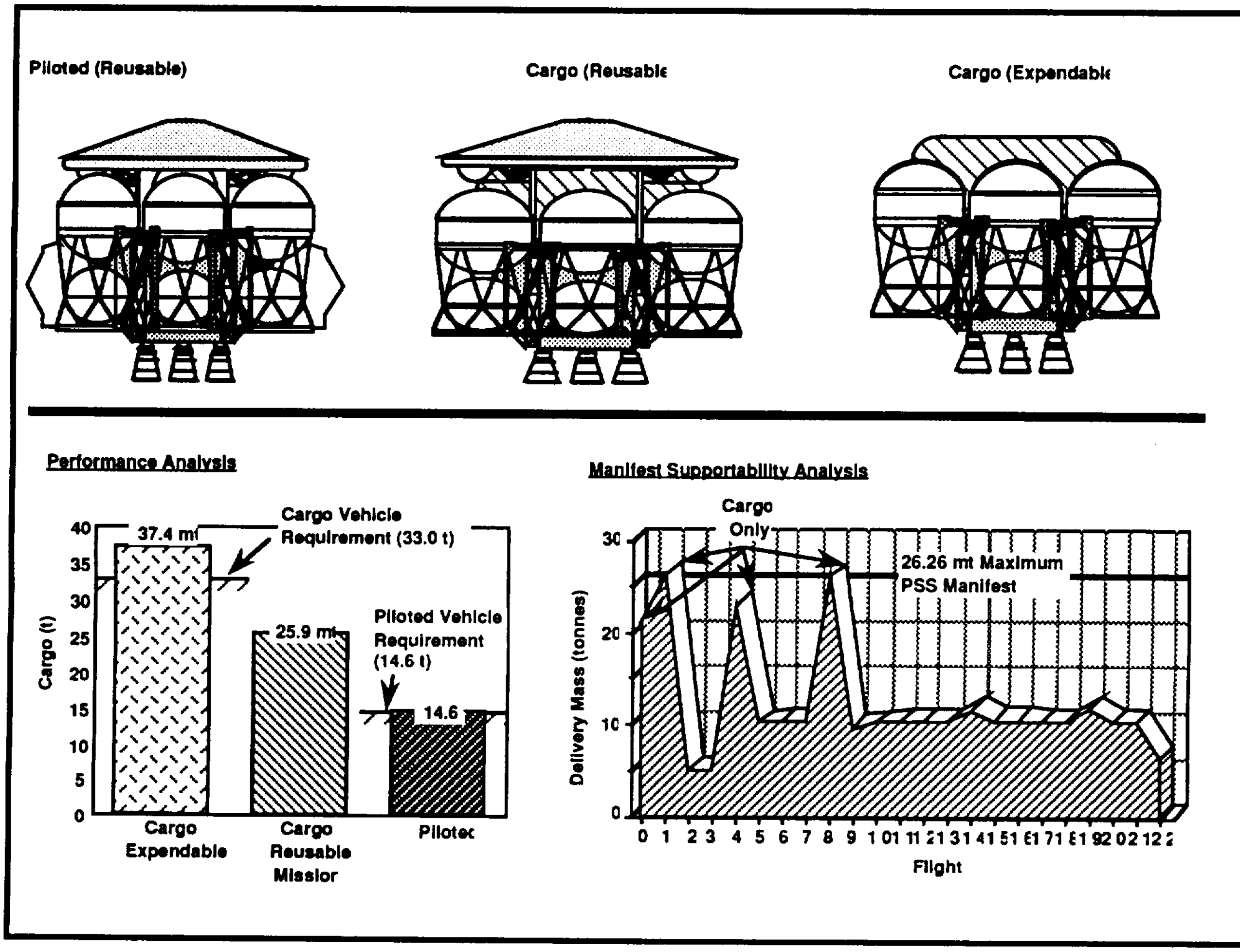
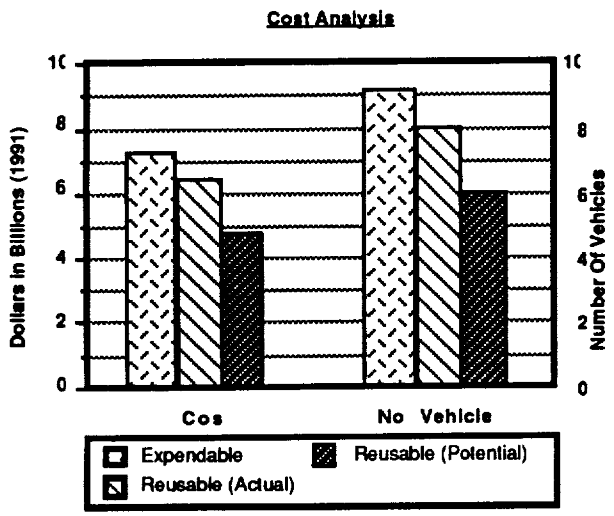


Figure 2.3.3.3-77 Reusability Trade Results

billion. By reallocating a small portion of cargo to one or more piloted missions, the two remaining cargo missions can be reused. With all three cargo missions flown in the reusable configuration, the vehicle cost savings increases from $0.8 to $2.4 billion. Reusing the cargo vehicles also provides the means for a final systems checkout prior to committing a crew to lunar launch.

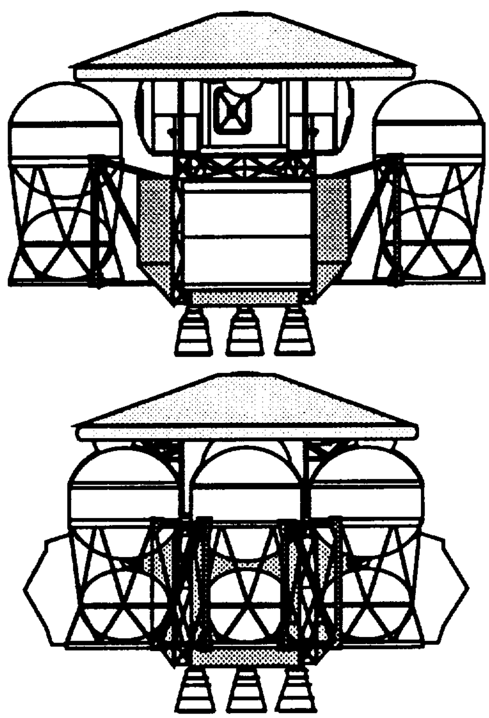
2.3.3.3.3.7 Final STV Configuration Recommendation—Figure 2.3.3.3-79 illustrates the configuration selected as a result of the final downselect process. The Single Propulsion System Family represents the best STV configuration that supports the Lunar design reference missions. Key attributes of this family include:

- Lowest LCC
- Lowest number of critical operational failure modes
- Meets all piloted and cargo only requirements, while featuring the highest cargo expendable capabilities.



- Three Cargo Only
- Mission Space-Based Reusable Piloted Configurations Yield a $.8 Billion Reduction in Vehicle Cost
- Reallocation Of Current PSS Manifested Cargo Could Provide an Additional $1.6 Billion in Savings
- Use of a Reusable Cargo System Prior to the First Piloted Mission, Provides Total System Checkout Capability

Figure 2.3.3.3-78 Economic Impact Of Reusability



- Configuration Derived From Initial Configuration Identification and Architectural Screen
- Low Life Cycle Cost
  - Single Propulsion System
  - Single Crew Module
  - High Reusability Of Elements
- Space Based
- Meets Piloted & Cargo Only Requirements With One System. Highest Expendable Cargo Capability.
- Lowest Number Of Mission Failure Modes
  - No Aerobrake Penetrations
  - No Cargo/Crew Transfer

Figure 2.3.3.3-79 Final Configuration Recommendation

2.3.3.3.4 Lunar LOX Utilization—The objective of this study was to make realistic decisions defining the extent to which vehicle, architecture and programmatic decisions are influenced by the expected existence of LLOX on the moon. In the conduct of this study a specific set of groundrules and assumptions were established to frame the scope of the analysis, Table 2.3.3.3-3. In general we have assumed that the nation will not be able to afford an initial set of launch and lunar transfer/landing vehicles and a follow-on set that is sized specifically to make use of LLOX. On the other hand, we cannot begin a transportation architecture based on LLOX because the first mission cannot be flown since there is no LLOX a priori.

Table 2.3.3.3-3 Groundrules and Assumptions

- Only One Class of HLLV
  - A New HLLV Is Not Developed To Be Optimum for the Lunar LOX Scenario
- No Changes or Modifications to the Lunar Vehicle Concepts
  - Determine the Feasibility of Lunar LOX When Applied to the Current Set of Vehicle Concepts and Architectures
- Lunar LOX Production Facility Produces Enough LOX To Meet the Needs of the Lunar Vehicles
- Lunar LOX Production Facility Costs Include:
  - Development
  - Delivery to the Lunar Surface
  - Setup
  - Operations & Maintenance
  - Delivery of the LOX to Low Lunar Orbit

Continuing with our assumptions, the LLOX facility will be able to produce as much LOX as needed for the lunar transportation system (LTS). (Needs of the lunar base will be small compared to the LTS.) In performing cost trades, the costs of LLOX must include research, development and construction, delivery to the lunar surface, setup on the lunar base (including EVA costs), annual operations costs (which include earth-based controllers, spares production, delivery and maintenance) and, finally, the cost of delivering the LLOX to any destination other than where it is produced on the lunar surface.

2.3.3.3.4.1 Cost , Performance, and Design Feasibility Analysis—To define the scope and direction of the study the logic flow (decision tree) was developed (Fig. 2.3.3.3-80). In this trade we assume that we want to use LLOX and then show the most cost effective means of utilizing LLOX. Based on the bottom line of this analysis, we can then go back and answer the top level question of developing LLOX. Given that we are to develop LLOX, the first question becomes, Should we change our vehicles so that we can optimize LLOX utilization or should we use vehicles that are not specifically designed for LLOX? The answer to this question is that we should use existing vehicles because if we bookkeep the cost of developing a new or modified launcher and LTS, we clearly cannot earn back tens of billions of dollars in lost capital based on earning rates shown on a subsequent chart.

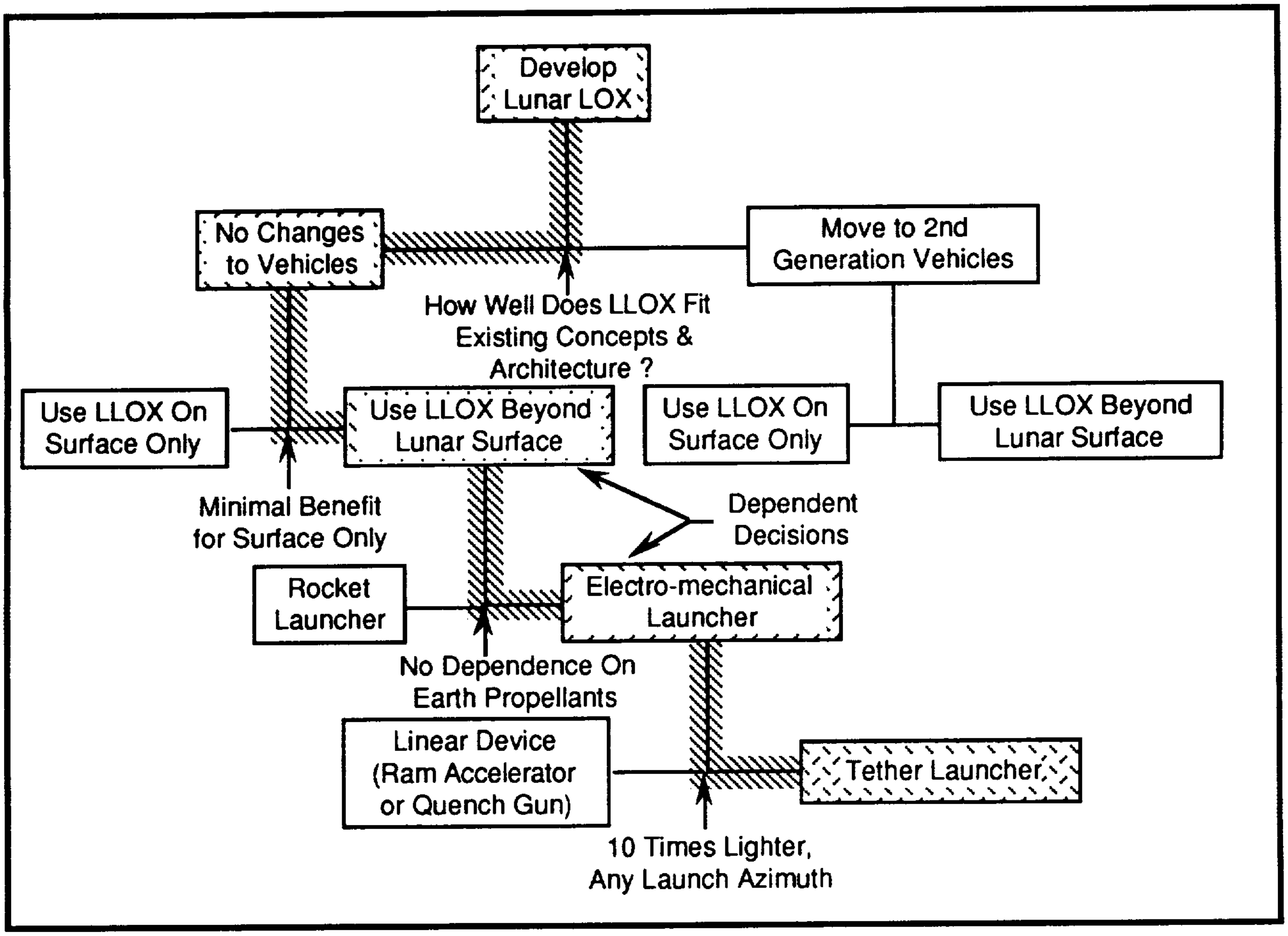


Figure 2.3.3.3-80 LLOX Decision Tree

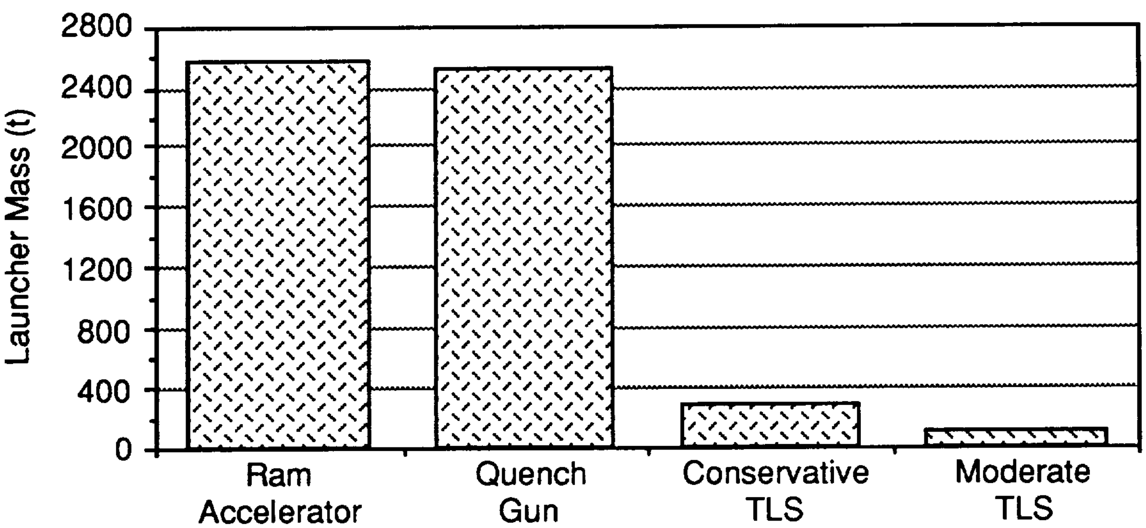
Given that we must use "first generation" vehicles that are not tuned for LLOX use, the next question becomes, How will we use LLOX in a first generation architecture, i.e. where will LLOX be made available for the LTS? The options are those node locations where a vehicle can accept LLOX. These include the lunar surface, LLO, a libration point or a low Earth orbit (LEO). Since the cost of LLOX will go up dramatically as it is moved away from where it was mined, the most economic points of use are those closest to the lunar surface. (i.e. the lunar surface and LLO). Taking LLOX beyond LLO requires new transportation elements (tankers) and has a significant dependence on Earth resources. It will be shown later that even LLO requires nearly one kilogram of propellants from Earth for every kilogram of LLOX lifted into orbit. However, if LLOX is used on the surface, the demand created by transportation vehicles needing ascent propellant is very small and will later be shown to be insufficient in paying back the investment. For LLOX to be cost competitive with Earth propellants it must simultaneously be in high demand and also independent of Earth resources on a recurring basis. These two facts, in parallel, drive the answers to two questions- where to use LLOX and how it gets to LLO. The demand for LLOX exists in LLO, where the LTV can be refueled for the trip to Earth and where the LEV can be loaded with descent oxidizer, but since using a rocket powered vehicle doesn't save any lift mass from Earth, we are forced to deliver it to LLO by an electro-mechanical device.

Given that we want to produce LLOX and have it available where there is a significant need, we must electro-mechanically deliver it to LLO. How best to do this? The answer, based on installation mass comparisons, is to use a rotating tether launcher. This device would use two tethers, one with the LLOX and the other with a regolith counter balance, to fling a drop tank of LLOX into a zero by 100 km lunar orbit where a timed motor would burn at apolune to circularize the orbit. The LTS would then rendezvous with three different tanks, attaching to them and either draining them (LTV) or descending with them (LEV) to where they can again be filled and thrown into LLO.

Having LLOX available only on the lunar surface severely restricts the amount that can be used by the LTS. The LTS would have used most of its oxidizer by the time it landed with a heavy cargo, and possibly a crew cab as the amount of LOX needed for descent is about 3 times that needed for ascent. One scenario would land the LTS at the surface with LLOX for ascent to LLO in preparation for the inbound phase of the mission. The problem with this is that the LTS arrives in LLO with only about half the quantity of LLOX it would need just for its upcoming descent, to say nothing of supporting TEI and EOI. Also the vehicle's landing system would require redesign for the heavier surface mass expected with full LOX tanks and the engine may fail to fulfill fail-op/fail-safe requirements with the heavier vehicle lift-off mass. Since the LTS is left requiring more LOX in LLO to complete the mission, the LOX must come from Earth on vehicles that were designed to carry full loads of LOX, but are now off-loaded. This would mean flying LTS vehicles inefficiently, off-loading cargo, and still paying for the full cost of a lunar mission. LLOX must be made available in LLO in whatever supplies the LTS demands to perform a full-up mission.

Assuming that LLOX must be delivered to LLO to find a "good market" for it, how do we get it there without using Earth propellants? One answer is to use an electro-mechanical launcher of some type. Using the LEV, modified as a tanker, requires 0.82 kilograms of Earth propellants ($H2/O2$ @ 6:1 mixture ratio) for every kilogram of LLOX delivered to LLO. The savings in launches due to having LLOX becomes marginal and little room is left to save money that was invested in the LLOX facility initially. The possibility of bringing LLOX back to LEO quickly falls apart because there must be a dedicated set of vehicles to do this task. This is because the amount of LLOX that must be lifted off the lunar surface is measured in hundreds of tonnes per mission to enable enough LLOX to be delivered, using an aerobrake, into LEO, and come back to the moon.

The moon is an excellent location for an electro-mechanical launcher due to its lack of atmosphere and low launch ΔV of 1700 m/s. Figure 2.3.3.3-81 shows why the tether launcher was selected as the system to deliver LLOX to LLO. The tether launcher is much lighter than a linear gas/combustion device or a linear electromagnetic accelerator. Another discriminator is the fact that a linear device is spread out over several kilometers of terrain, driving up maintenance costs and limiting launch azimuth to only one direction. The tether launcher is compact, easily serviceable (entire tether can be replaced with a crane), and can launch a payload into any azimuth. The figure shows that the tether launcher is an order of magnitude less massive than linear devices, and reasonable advances in tether characteristic velocity (specific strength) rapidly drive the mass down further (see the moderate mass estimate which is only 10 times more massive than the payload it



- All Launchers Launch 10 tonne Payloads to LLO
- Ram Accelerator and Quench Guns are an Order of Magnitude Heavier
- Selection of Safety Factor (2 vs 1.4) and Cord Specific Strength (0.76 vs 0.5) Make A Factor of 3 Difference in TLS Mass (300 t vs 100 t)

Figure 2.3.3.3-81 Comparison of Tether Launcher to Other Systems would be launching). The tether launcher uses two 7 kilometer tethers, has a height of about 100 meters, and puts a radial acceleration of about 42 Earth-Gs on the payload.

2.3.3.3.4.2 Summary and Recommendations—Figure 2.3.3.3-82 shows the economics of lunar LOX. The two lines emanating from the $5000 million point represent the investment in the LLOX facility. The lines emanating from the origin are the revenue or earnings lines. Where they meet in time is the break-even point. Two lines emanate from each point, the upper line being discounted money and the lower being straight, constant value money. The discounted rate for the upper lines is a net 2% which is the discount rate minus the inflation rate. In other words it's the rate at which a dollar appreciates when invested. For discounted funds the break even point is in 13.1 years and for constant dollars 14.7 years, meaning the LLOX facility and tether launcher must endure for this period of time just to break even. If they do not, then a replacement set of facilities must be delivered before the savings in using LLOX displace the invested cost.

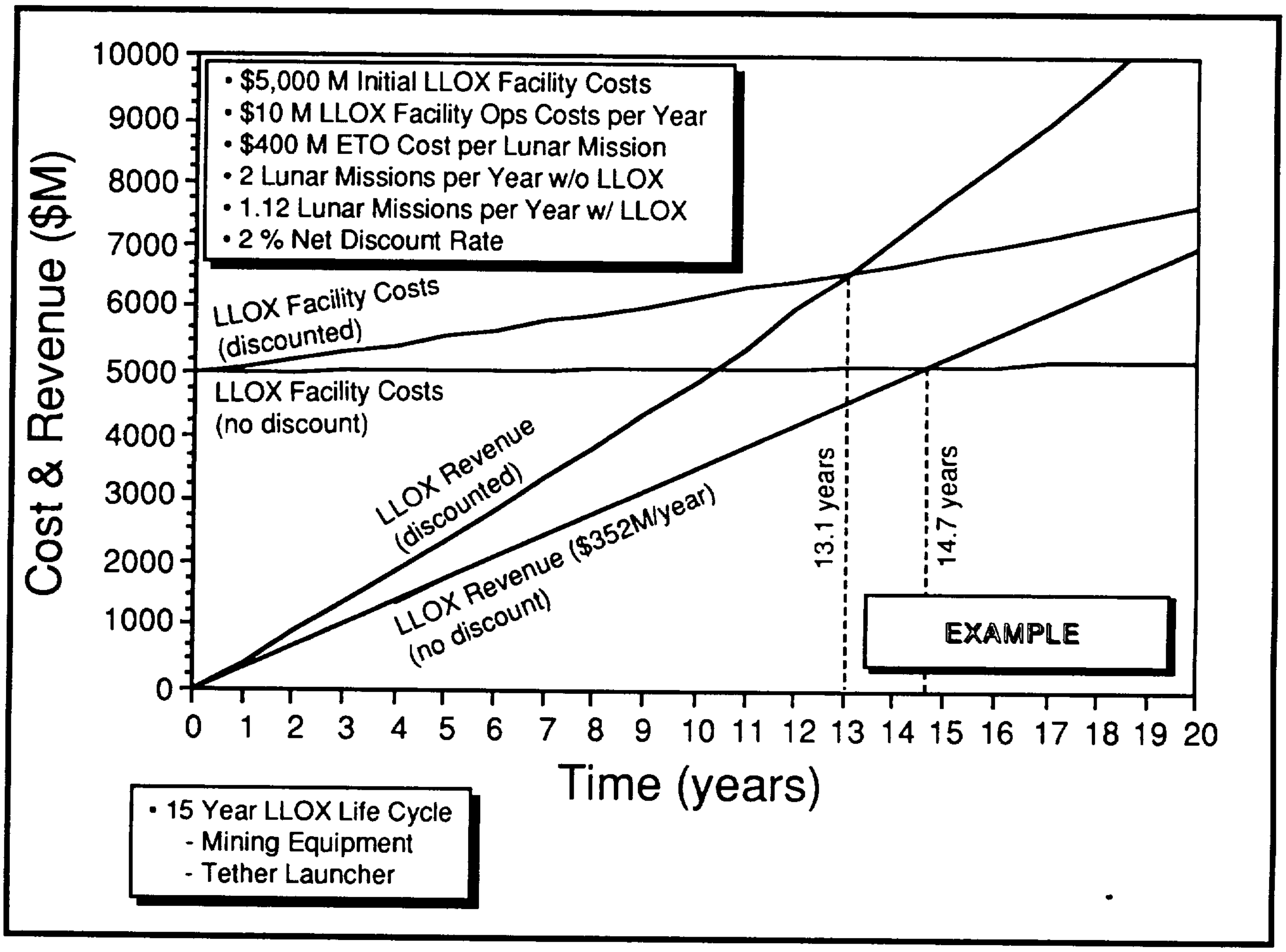


Figure 2.3.3.3-82 Lunar LOX Economics

The revenues come from a reduction in missions flown. We assume a mission costs $400M and that two missions are flown per year. With LLOX available in LLO and on the lunar surface, we can reduce the number of lunar missions per year from 2 to 1.12. Hence the savings is $352M per year. This savings would be considered revenue earned by LLOX. The $5B initial cost must include all aspects of producing LLOX and delivering it to LLO as stated in the assumptions and groundrules chart before. Assuming a $10M/year operations cost, which is barely discernible compared to the time value of money, this chart shows typical LLOX cost effectiveness.

In conclusion, it is recommended that the LLOX trade study be suspended until two key pieces of data are firmly in hand. One is the cost per kilogram of launching mass to low Earth orbit using an HLLV; and the second is the production cost of a pilot LLOX plant operating on the lunar surface. LLOX is a second generation surface activity and, therefore, should not be addressed until the first generation is implemented, or at least well underway. Key inputs into whether LLOX would be profitable are the cost of goods in LEO and the cost of LOX production on the moon. Trade studies at this point in time can assume many factors biasing the results to support a desired position. It is essential that actual data be inserted into the equation before investing billions of dollars in second generation activities on the lunar surface.

2.3.3.4 SPACE STATION SENSITIVITIES—

2.3.3.4.1 STV Mass Sensitivity Analysis—

2.3.3.4.1.1 Impacts to SSF Guidance, Navigation & Control—This analysis has assumed that a high-mass LTS is supported in a 15.3 x15.3 m servicing enclosure positioned on a lower keel of the Space Station. This configuration, derived from the November 1989 NASA 90-day study on Human Exploration, recommended the addition of a lower keel to support lunar operations.

Space Station Freedom flies at Torque Equilibrium Attitude (TEA), where aerodynamic and gravity gradient torques cancel. Current analysis indicates that the TEA of the Assembly Complete Station has a large negative pitch angle and will not meet the requirement to fly within +/- 5 degrees of Local Vertical, Local Horizontal (LVLH). The addition of a lower keel will significantly improve the pitch attitude (see Figure 2.3.3.4.1.1-1). Pitch and yaw attitudes are further reduced toward LVLH as the mass of the LTS is increased. Roll TEA attitude increases with additional LTS mass. However, over the range of potential LTS mass to be supported, station TEA will remain within the +/- 5 degree requirement.

LVLH as the mass of the LTS is increased. Roll TEA attitude increases with additional LTS mass. However, over the range of potential LTS mass to be supported, station TEA will remain within the +/- 5 degree requirement.

Baseline momentum storage capacity for SSF can be provided by a pallet containing 6 Control Moment Gyros (CMGs). Each CMG provides 3500 ft-lb/s of

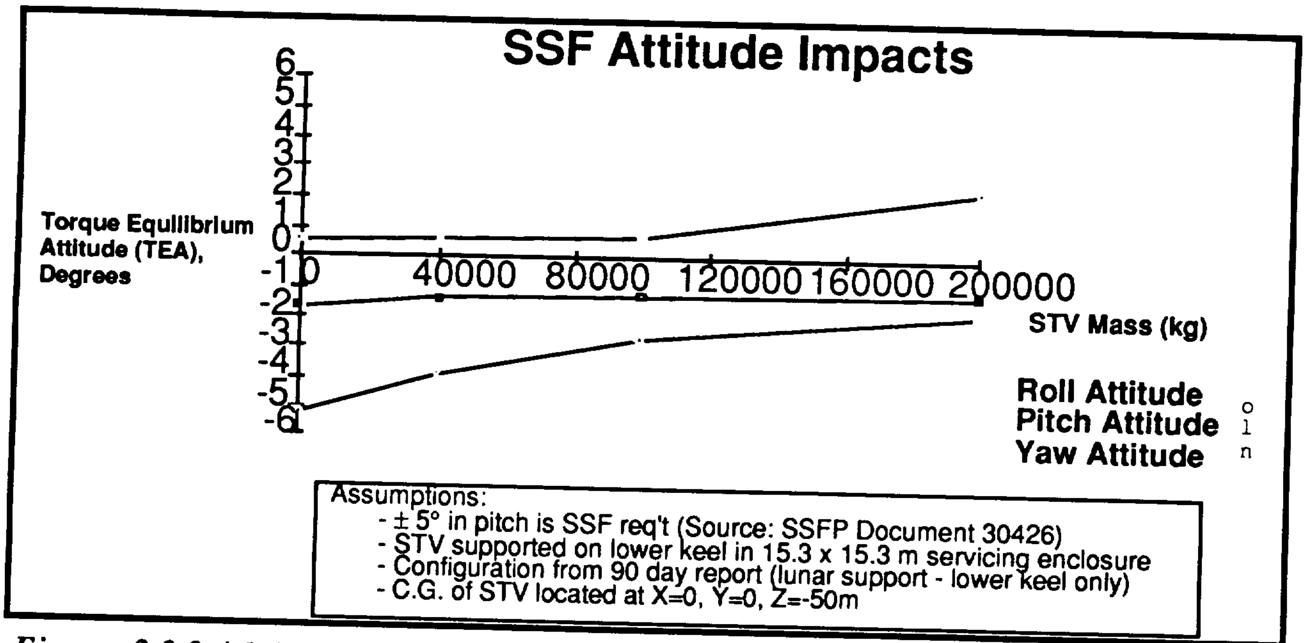


Figure 2.3.3.4.1.1-1 SSF Attitude Impacts

momentum storage for a total of 21000 ft-lb/s capacity at assembly complete. Required momentum storage capacity is a function of many variables including specific configuration and momentum management scheme during flight. Analysis using a momentum-management simulation indicates that increased LTS mass will have low impact on station control. Required momentum storage capacity initially increases then is reduced for a higher-mass LTS when the aerodynamic torque effects are offset by the large gravity gradient torque gains. Figure 2.3.3.4.1.1-2 shows that the maximum momentum storage requirements can probably be met by the addition of two or three CMGs over the range of LTS mass to be supported on a lower keel. Location of these additional CMGs is not critical and could be supported on or near the existing CMG pallet.

2.3.3.4.1.2 Impact on Micro g Users—LTS mass on a lower keel has a severe impact on the SSF microgravity environment. Even with an empty servicing enclosure, SSF cg would be below the desired centerline for the laboratory modules. In Figure 2.3.3.4.1.2-1, the microgravity contour

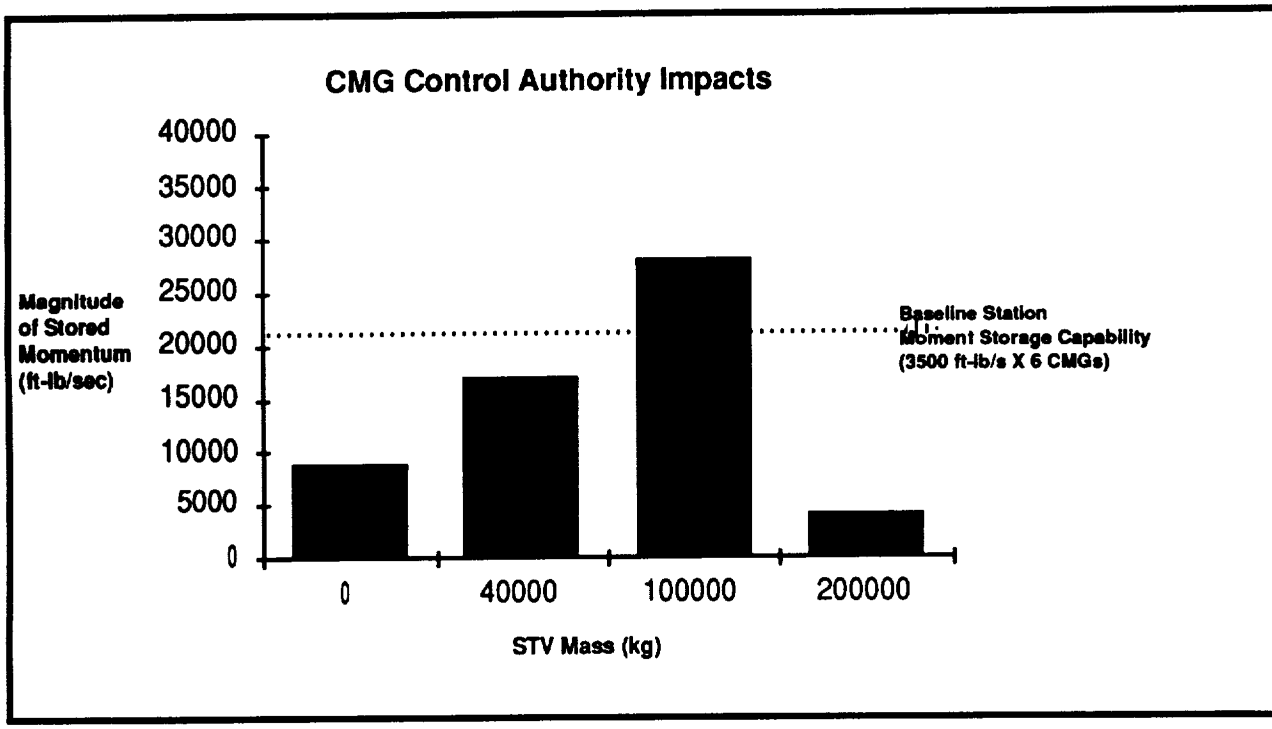


Figure 2.3.3.4.1.1-2 CMG Control Authority Impacts

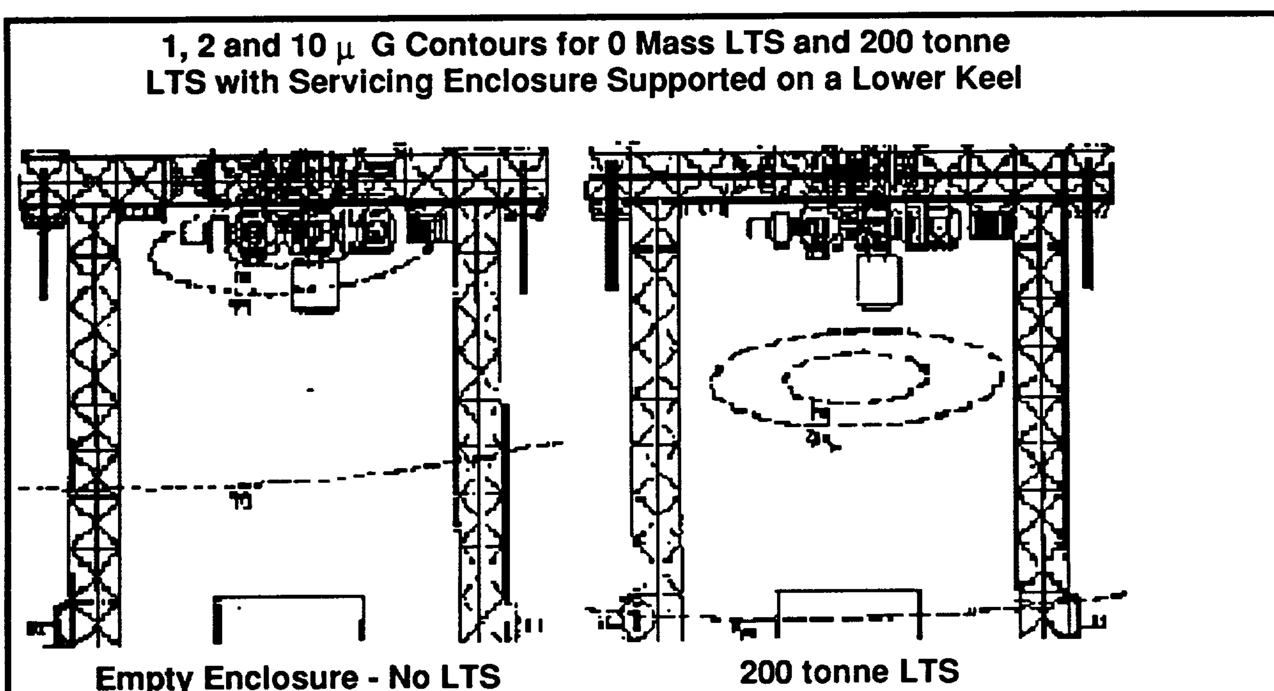


Figure 2.3.3.4.1.2-1 SSF Microgravity Environment

lines for 1, 2 and 10 microgravity levels for the 200 tonne LTS show that the additional mass significantly lowers the cg of the SSF, pulling the microgravity contours down. The addition of an upper boom at the same time as a lower boom ("dual keel configuration") will improve SSF microgravity environment, offsetting the lowered center of gravity. SSF center of gravity location is shown as a function of LTS mass. A Level II directive (BB000610A) was recently issued changing the previous requirement of 10 micro-g in the laboratory modules. This directive states that the station "shall be capable of providing quasi-steady acceleration levels not to exceed 1 mg for at least 50% of the user accommodation locations in each of the pressurized laboratories (US Lab, ESA and JEM PM at AC)." As shown in the plot of percentage total laboratory volume within 1 and 10 microgravity levels (Fig. 2.3.3.4.1.2-2), any appreciable mass LTS supported on a lower keel will not be able to meet this directive.

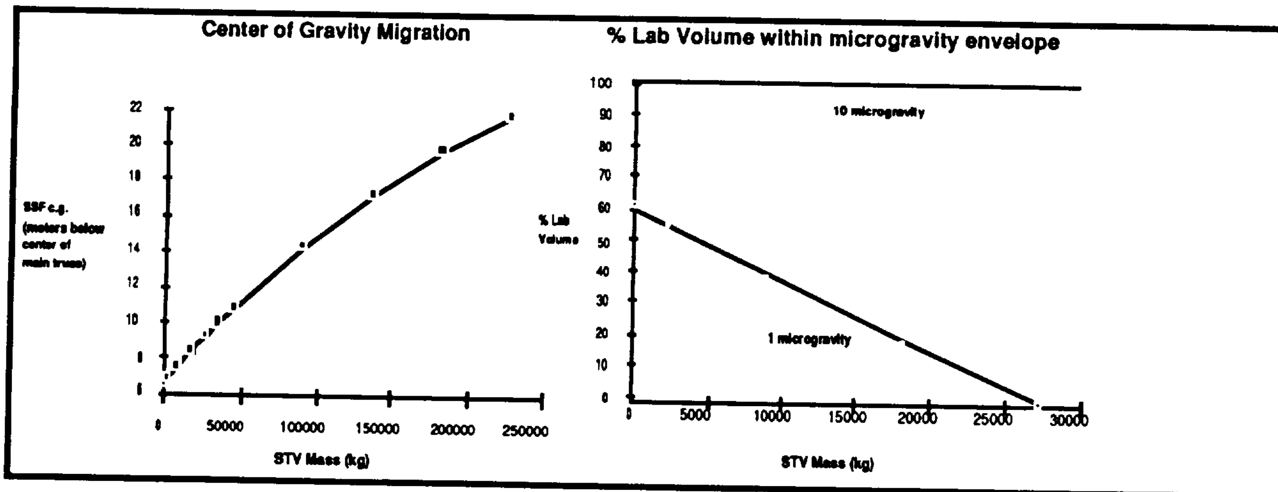


Figure 2.3.3.4.1.2-2 SSF Microgravity Environment Sensitivity

2.3.3.4.1.3 Impact on Reboost Logistics—Reboost propellant required during a low solar cycle year is shown as a function of LTS mass. Figure 2.3.3.4.1.3-1 compares the propellant required for a low-mass LTS based on the main truss as an attached payload with a large-mass LTS supported on a lower keel. The addition of the lower keel and servicing enclosure increases station propellant use about 5000 pounds of hydrazine. After this initial increase, the entire range of LTS mass will not require more than one additional propulsion module (8000 pounds of hydrazine) for the low solar cycle year.

Yearly required reboost hydrazine is shown for both low and high solar cycle years over the range of LTS mass on a lower keel. The high solar cycle year is the worst-case for reboost requirements and will require up to two additional propulsion modules over the LTS mass range.

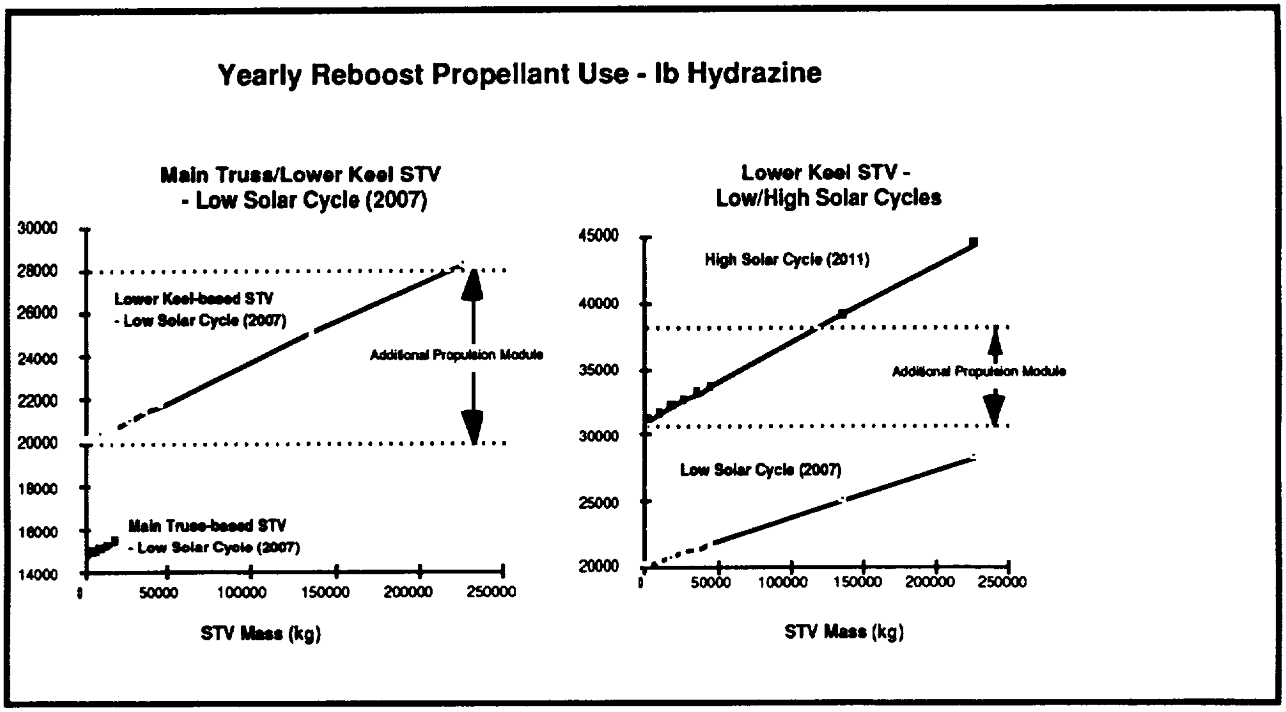


Figure 2.3.3.4.1.3-1 SSF Reboost Logistics

2.3.3.4.2 LTS Size Sensitivity Analysis—

2.3.3.4.2.1 Enclosure Size & Location—The size to which an LTS could grow within the constraints of the space station system is governed by limits applied to the size of its enclosure. The two dimensional constraints are in the Y (or latitudinal) dimension and the Z (or radial) dimension of the station configuration. The LTS enclosure is assumed to be placed in a location bounded by a "lower keel," or two downward pointing extensions of the truss structure connected by a cross boom. The boom dimensions are governed by the physical space available on the main truss structure as well as constraints in station controllability that govern the extent to which the truss can grow downward.

As depicted on the Figure 2.3.3.4.2.1-1, the maximum amount by which the enclosure can grow along the Y axis is 35 meters. Thus the maximum LTS diameter within the enclosure will be 31-33 meters, depending on safety factors. The limit, in the Z dimension, has two components. Forward of the lower keel truss structure plane, the maximum enclosure growth limit is 26.6 meters due to clearance requirements for LTS docking to the space station. Aft of the truss structure plane, the limit is relaxed to 43.8 meters, which is bounded by the envelope for a pressurized logistics module attached to a min-node.

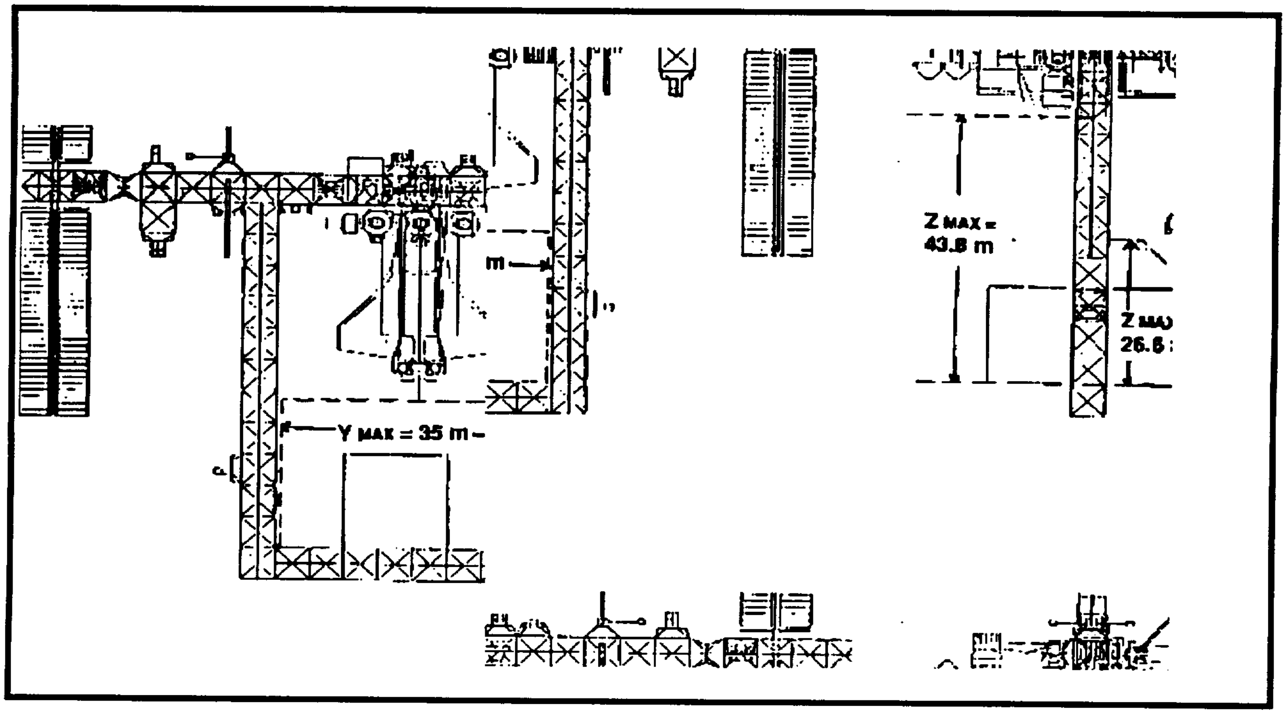


Figure 2.3.3.4.2.1-1 Enclosure Limits

2.3.3.4.2.2 Impacts to Reboost Logistics & Micro g Users—As the size of the LTS enclosure increases, there are also impacts to space station reboost logistics planning and the station microgravity environment. As the frontal area of the enclosure grows, the drag coefficient increases, and extra propellant must be provided to the space station for altitude maintenance. The SSF reboost propulsion system is based on a monopropellant hydrazine system that is resupplied by propellant modules that contain 8000 pounds each. Four of these pallets per year are planned for delivery to the station. As shown on the left side of Figure 2.3.3.4.2.2-1, even when the enclosure reaches its maximum size of 35 x 35 meters, less than one additional propellant module would be needed in a high solar-cycle year. This occurs when reboost requirements are at a maximum due to atmospheric expansion.

As the enclosure size grows, added drag and mass cause the station center of gravity (and microgravity ellipses) to move lower relative to the experiment module section. This movement, less than three meters from minimum to maximum enclosure size, can be considered a minimum impact.

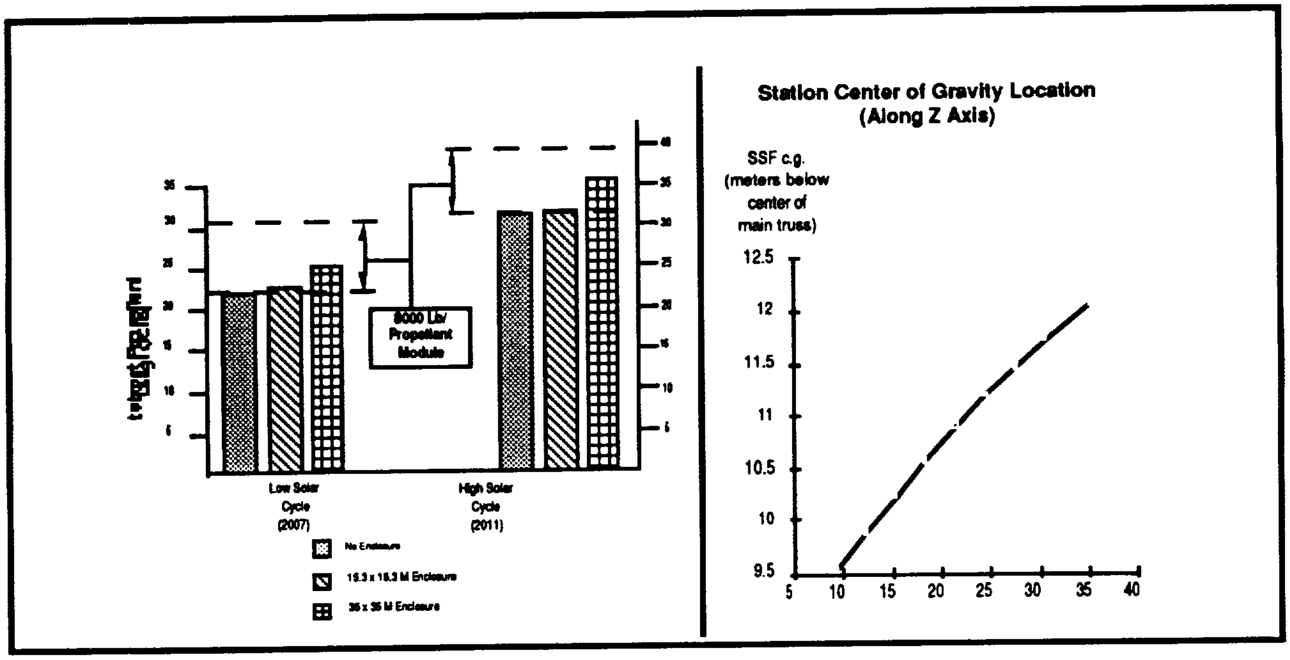
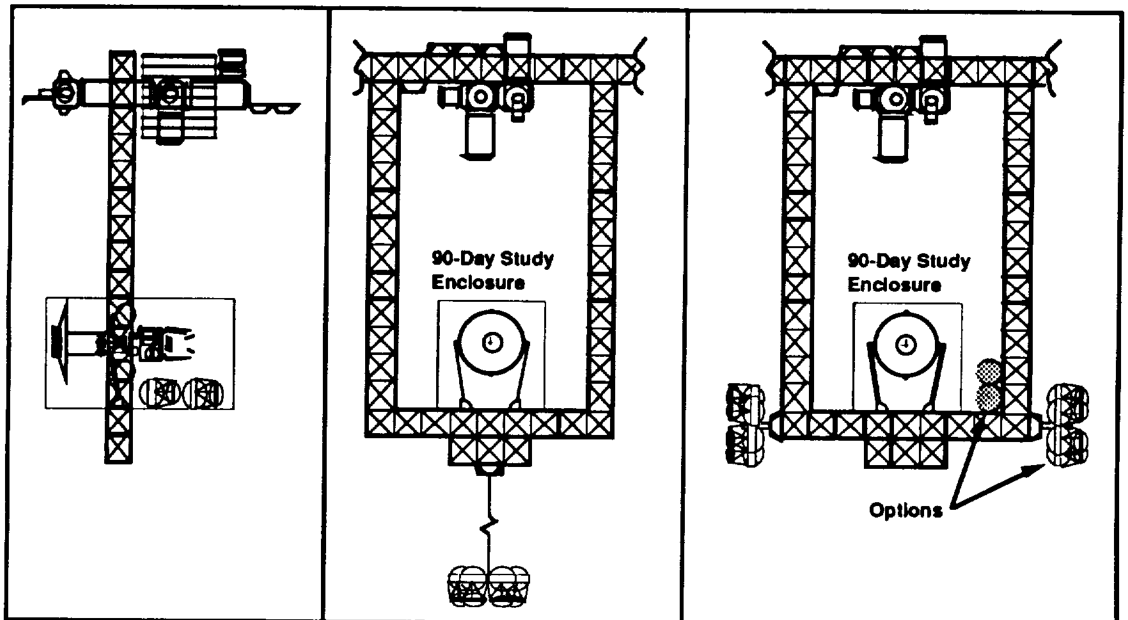


Figure 2.3.3.4.2.2-1 ***SSF Reboost Logistics & Microgravity***

2.3.3.4.2.3 Propellant Management Trade—One of the key concerns with LTS accommodations at SSF is where to store the LTS propellant tanks after they are received at SSF and before assembly with the LTS. In conjunction with this trade study, three options were identified as potential locations for the LTS propellant tanks (Figure 2.3.3.4.2.3-1). The first option was to mount the propellant tanks within the SSF servicing enclosure. The second was to mount the propellant tanks on a tether away from SSF. The third was to mount the propellant tanks elsewhere on the lower keel outside the servicing enclosure.

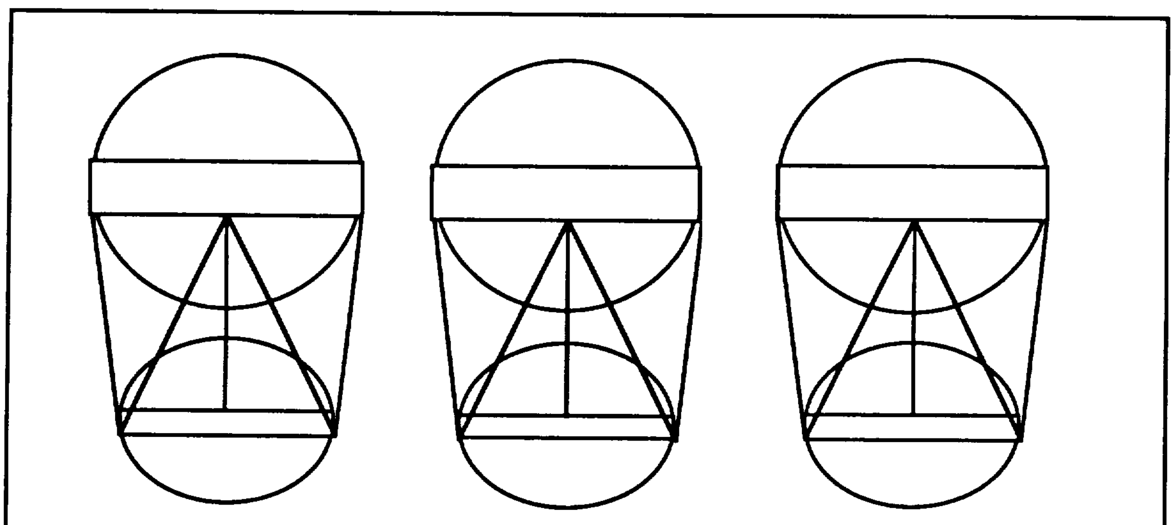
The Lunar Transfer Vehicle propellant tank configuration shown in Figure 2.3.3.4.2.3-2 consists of six equal size tanksets for a total propellant quantity of 156 tonnes (343,000 pounds). The LH2 tanks are 15 feet in diameter and 14.52 feet long. The LO2 tanks are 12 feet in diameter and 9.22 feet long.

The Lunar Transfer Vehicle propellant tank configuration is shown in Figure 2.3.3.4.2.3-3. It consists of six tanksets of equal size for a total propellant quantity of 156 MT (343,000 lbs). The LH2 tanks are 15 ft in diameter and 14.52 ft long. The LO2 tanks are 12 ft in diameter and 9.22 ft long.



OPTION 1	OPTION 2	OPTION 3
Mount propellant tanks within enclosure	Mount propellant tanks on tether	Mount propellant tanks on lower keel

Figure 2.3.3.4.2.3-1 Propellant Storage Location Options



- 6 tanksets of equal size
- Propellant quantity of 26 MT/set
- LH2 tank is 15 ft dia., 14.52 ft long
- LO2 tank is 12 ft dia., 9.22 ft long

Figure 2.3.3.4.2.3-2 Lunar Transfer Vehicle Propellant Tank Configuration

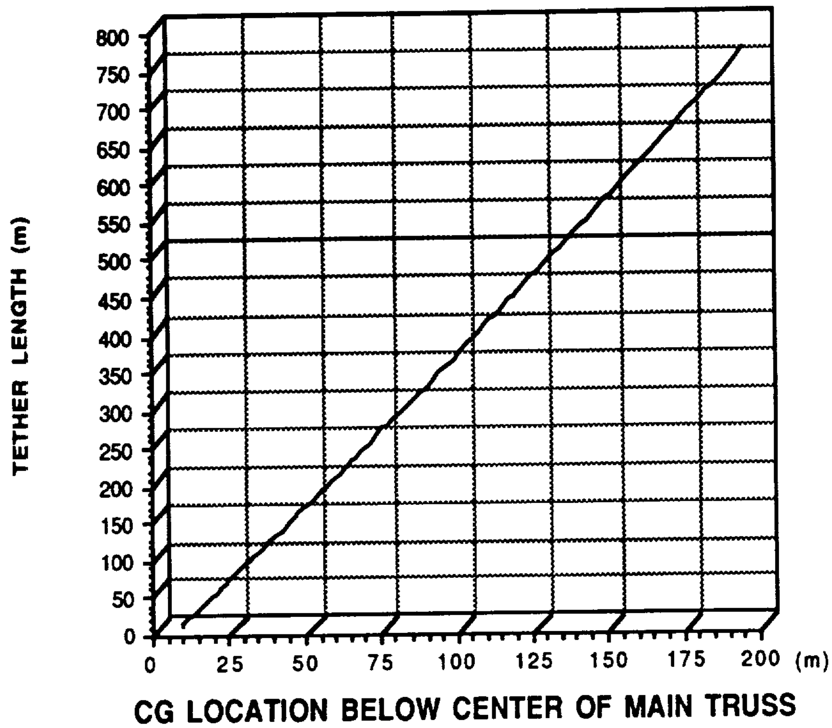


Figure 2.3.3.4.2.3-3 Tether Length Versus Station Center of Gravity

2.3.3.4.2.3.1 Effects of Thermally Insulated Enclosure on Propellant Tanks—If the propellant tanks are enclosed in a thermally insulated enclosure (e.g., the NASA 90-day Study servicing facility enclosure) then the required Multi-Layer thermal Insulation (MLI) on the propellant tanks could be reduced. The decrease to the recurring launch weight of the propellant tanks would also require a trade against the one-time launch weight of the thermally insulated servicing enclosure.

The technical approach to this analysis was to estimate the reduction in propellant tank MLI thickness versus enclosure MLI thickness. This assumes both MLIs have identical thermal characteristics and no heat loss through enclosure openings.

The results indicated that the weight of the servicing enclosure MLI is substantially higher than the weight saved on the reduced tank MLI. Reduction in MLI thickness on the propellant tanks should only be considered if the servicing enclosure is to be insulated for other reasons.

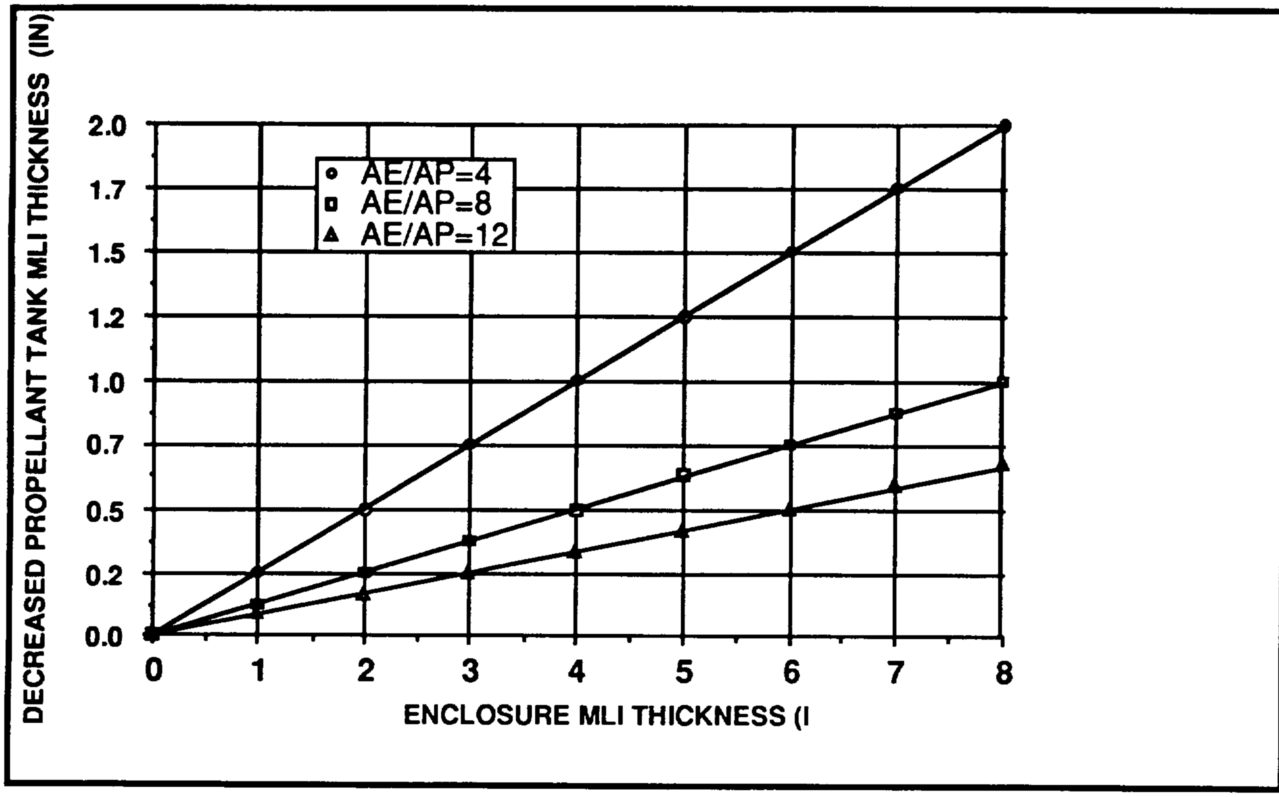


Figure 2.3.3.4.2.3.1-1 Decreased Propellant Tank MLI Thickness Versus Enclosure MLI Thickness for Various Enclosure Surface Area to Propellant Tank Surface Area Ratios (AE/AP)

Figure 2.3.3.4.2.3.1-1 presents a plot showing decreased propellant tank MLI thickness versus enclosure MLI thickness for various enclosure surface areas to propellant tank surface area. The following assumptions were used in generating this plot: (1) MLI for both propellant tank and enclosure have identical thermal qualities; (2) MLI thermal conductivity is not a function of MLI temperature; and (3) negligible heat loss from enclosure through openings. The plot also assumes that heat loss is proportional to MLI conductivity, temperature, and surface area and inversely proportional to MLI thickness.

2.3.3.4.2.3.2 Impacts of Ruptured Hydrogen or Oxygen Propellant Tank on SSF's Environment—As part of the trade to determine the appropriate location for storing LTS propellant tanks, the contamination effects of ruptured hydrogen (H_2) and oxygen (O_2) tanks were analyzed. Although the physical impact of exploding debris on station structure and systems is potentially more severe than the contamination only from the H_2 and O_2, the exploding debris impact has not been analyzed. The risks and safety hazards associated with debris caused by exploding propellant tanks has been left to future analyses.

The key issues associated with these contamination analyses are the length of the tether to minimize the effect on the environment if tanks fail, and the effects on the environment if tanks located on the lower keel fail. There may be a need for shields to minimize effects on station environment.

The technical approach to this analysis was to calculate propellant gas density profiles versus distance from failed propellant tank based on spherical source flow expansion.

The results indicate that perturbation of station environment is high for most failure modes if failed propellant tanks are near the station. It is recommended that propellant tanks be tethered away from the station. If tanks were to be located on the lower keel, shields would need to be incorporated to protect the station environment from the potential of a ruptured H_2 or O_2 tank.

Figures 2.3.3.4.2.3.2-1 and 2.3.3.4.2.3.2-2 present plots estimating effects on station environment (density profiles) for various failure modes (propellant lost from tank) and tank locations for H_2 and O_2 tanks respectively. The following assumptions were used in generating this plot, (1) the expelled propellant from a failed tank is assumed to expand radially and uniformly from failure location, (2) H_2 and/or O_2 flashes to gas phase are based on a tank pressure of 15 psia, (3) gas velocity is based on sonic expansion at boiling temperatures of 36° R for hydrogen and 162° R for oxygen, (4) propellant tank failure and expulsion of gas is assumed to occur in 1.0 second, and (5) propellant expansion density profiles were calculated based on simple source flow approximation.

2.3.3.4.3 STV Assembly Sensitivities—The primary method used to analyze the sensitivity of LTS configurations to assembly operations was to examine the postulated space station based assembly support hardware, and determine if any LTS assembly operations would not be supported by this hardware. In this analysis, baseline space station mechanical devices were examined for their applicability to LTS assembly.

As depicted in Figure 2.3.3.4.3-1, the baseline space station has at least three mechanical systems that may be adapted to the LTS program. These devices are the mobile servicing system (mobile transporter and space station remote manipulator system), the unpressurized docking adapter, and the capture latches used for attaching the unpressurized logistics carriers and propulsion modules to the baseline space station integrated truss assembly.

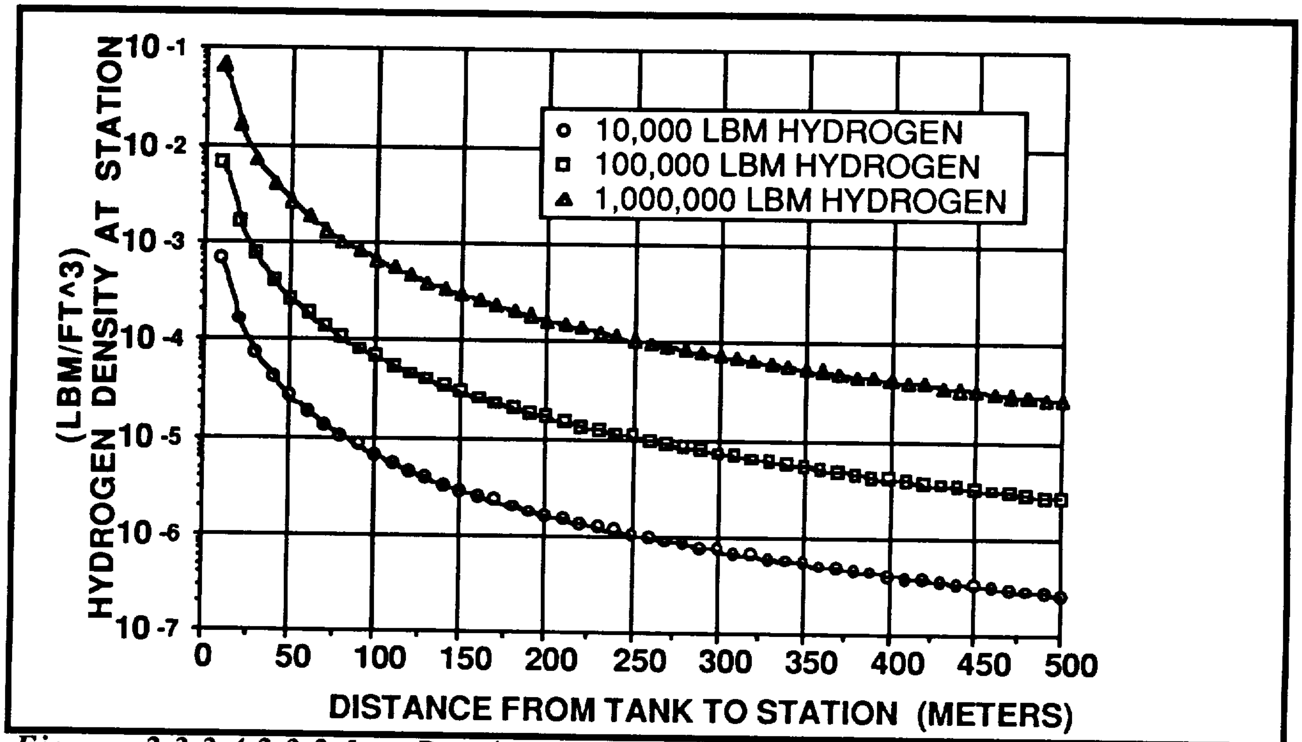


Figure 2.3.3.4.2.3.2-1 Density at Station Versus Distance from Ruptured Hydrogen Tank

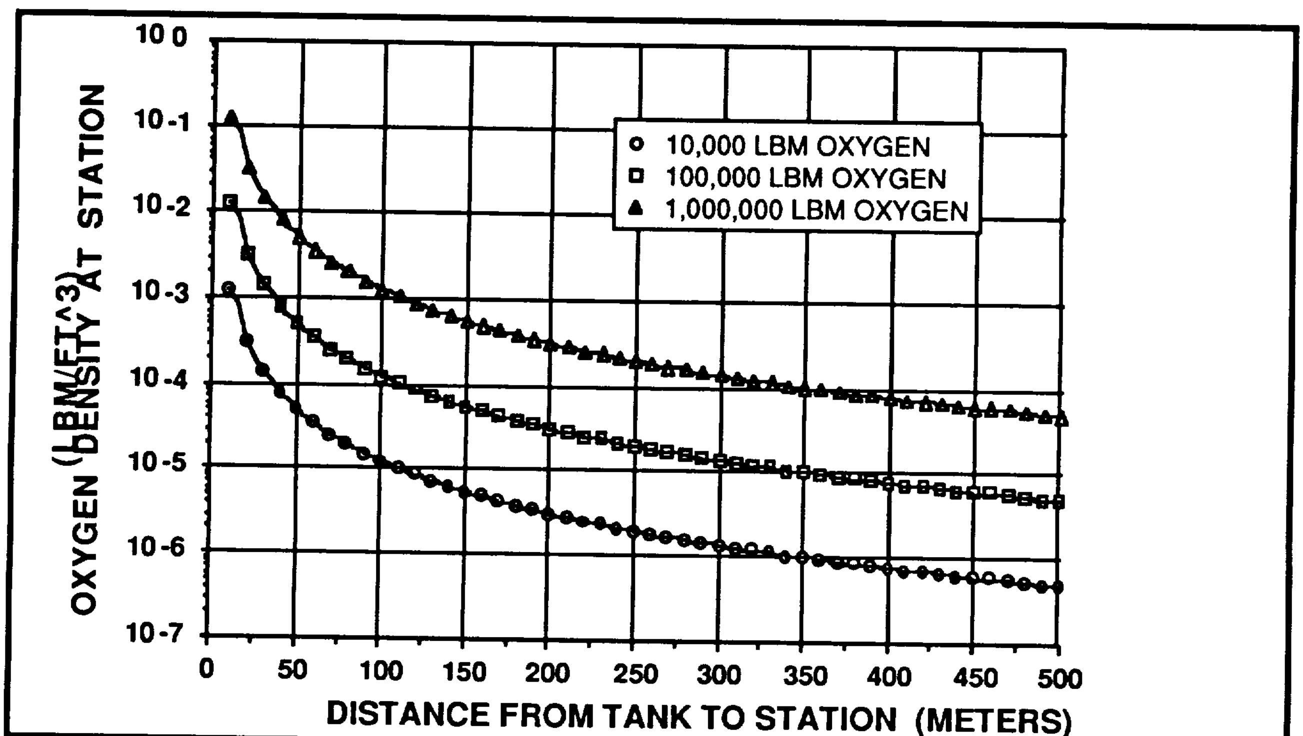
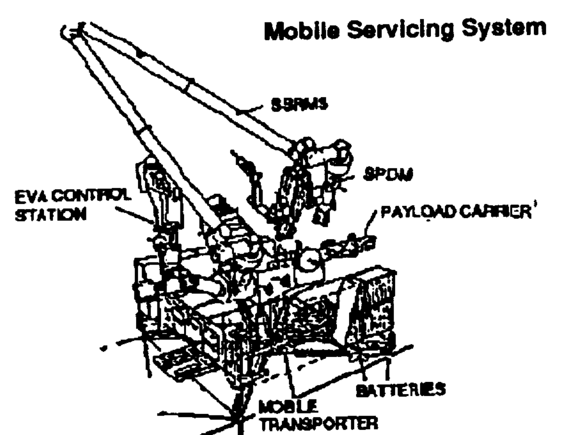
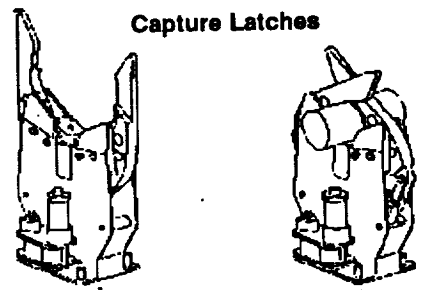
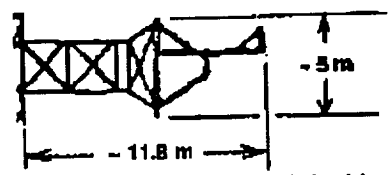


Figure 2.3.3.4.2.3.2-2 Density at Station Versus Distance from Ruptured Oxygen Tank





Unpressurized Docking Adapter



Mobile Servicing System Maximum Payload Capabilities

Station Configuration	Mass (kg)	Diameter (m)	Length (m)
Assembly & Baseline	20,900	4.5	17.0
Growth **	128,000	13.4	44.7

Figure 2.3.3.4.3-1 Space Station Mechanical Devices to Support STV Assembly

The unpressurized docking adapter may be modified to allow the LTS or portions of it to dock with the station. The capture latches, which are sized to accommodate either 3.00 inch or 3.25 inch STS payload trunnion pins, may be well suited for mounting LTS cargo elements to the truss structure before and during the assembly process. The mobile servicing center or some derivative of it is necessary for the performance of the LTS assembly functions. The Mobile Servicing System (MSS) can accommodate payloads up to 128 tonnes. This should cover the full range of LTS components, with the exception of a fully fueled lunar transfer vehicle. Therefore, the design of the LTS vehicle supports the use of space station baseline mechanical devices.

Although a number of SSF mechanical systems can be adapted for use in the LTS program, there are still several mechanical systems required for the LEO servicing facility that will be unique to the LTS program. These include an LTS core stage handling fixture, engine removal support hardware, LTS stack deployment device, and enclosure opening and closing mechanism. These devices will have to be more clearly defined so that their functions and operational complexity may be better determined.

Concerning current SSF mechanical devices that can be adapted to the LTS program such as the Space Station Remote Manipulator System (SSRMS), the STS docking adapter, and the SSF capture latches, more analysis will have to be performed to determine the degree to which these satisfy the LTS mission without modification, and what modifications would have to be made to completely satisfy LTS operations.

For the SSRMS, the issue exists as to whether a dedicated unit is required for LTS assembly and operations, or whether the SSF baselined unit can satisfy both LTS assembly and SSF housekeeping and payload requirements and timelines. Also there is the potential impact of dynamic loads on the SSRMS due to propellant sloshing in the propellant tanks and how the SSRMS will translate into and out of the LEO servicing facility enclosure.

Another potential impact on current SSF mechanical devices is whether the STS docking adapter needs to be upgraded for LTS operations. Coincidentally, if the LTS wants to take advantage of a STS docking adapter, this feature would have to be built into the design. Finally, if SSF capture latches are to be used, the ETO trunnions would have to be designed to be compatible.

2.3.3.4.4 Power Usage Sensitivity Analyses—The baselined assembly complete space station provides a maximum of 75 kW from four photovoltaic power modules. This 75 kW of power is split between station housekeeping and station user payload power. There is no surplus of power in the Phase I SSF. Therefore, if an LTS servicing facility or any other evolutionary function is required post-assembly complete, additional power modules will have to be added to supply the additional power. The current growth path for SSF uses pairs of Solar Dynamic (SD) power modules for 50 kW increments. Due to the life cycle cost, lower power degradation, lower resupply, and lower drag of SD over photovoltaic power, if less than 50 kW additional power is required in an evolutionary station, there could be the option of adding two photovoltaic power modules for an additional 37.5 kW.

Power to support the presence of an LTS and LEO servicing facility only can be accommodated with 37.5 kW additional for LTS powers up to 12 kW. This includes approximately 10 kW for the servicing facility and 10 kW if additional crew facilities are required (Figure 2.3.3.4.4-1). These values are based on the "On-Orbit Assembly/Servicing Task Definition Study" conducted by McDonnell Douglas in 1989, and are rough averages. Peak power demands and connected power could vary.

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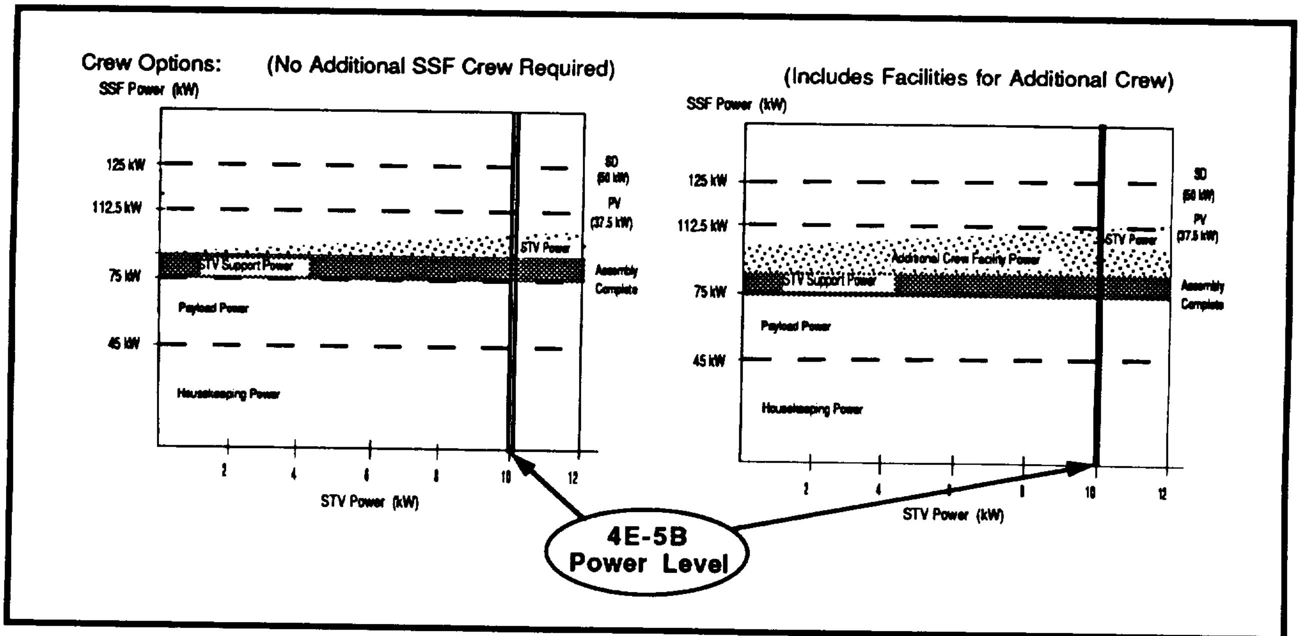

Figure 2.3.3.4.4-1 Impacts of Providing Power to 4E-5B LTS Configuration

2.3.4 Subsystem Analysis

With the completion of the system level and mission studies as well as a LTS configuration recommendation, there existed sufficient data to support a detailed and comprehensive study and analysis activity in the subsystems that make up the LTS. Through the configuration analysis effort, three key subsystems were identified; avionics, propulsion, and aerobrake. The avionics subsystem analysis addressed power, weight, built-in-test equipment, and technology issues with a goal to provide significant program pay-offs. Propulsion studies addressed primary propulsion and reaction control issues as well as utilization of the propellant to support power and life support systems. Aerobrake studies focused around materials, design, and operational issues. Figure 2.3.4-1 shows the relationship the subsystem analysis activities have with the overall study and analysis task.

2.3.4.1 **Avionics Analysis**—The following is a road-map of the derivation of several key avionics requirements to meet the STV design reference missions. Three distinct classes of requirements become evident from this task analysis: 1) cargo type, 2) mission duration, and 3) reusability. In addition to these, there is a second set of requirements derived from the particular launch system used to place the LTS/STV elements into earth orbit (manned or unmanned launch

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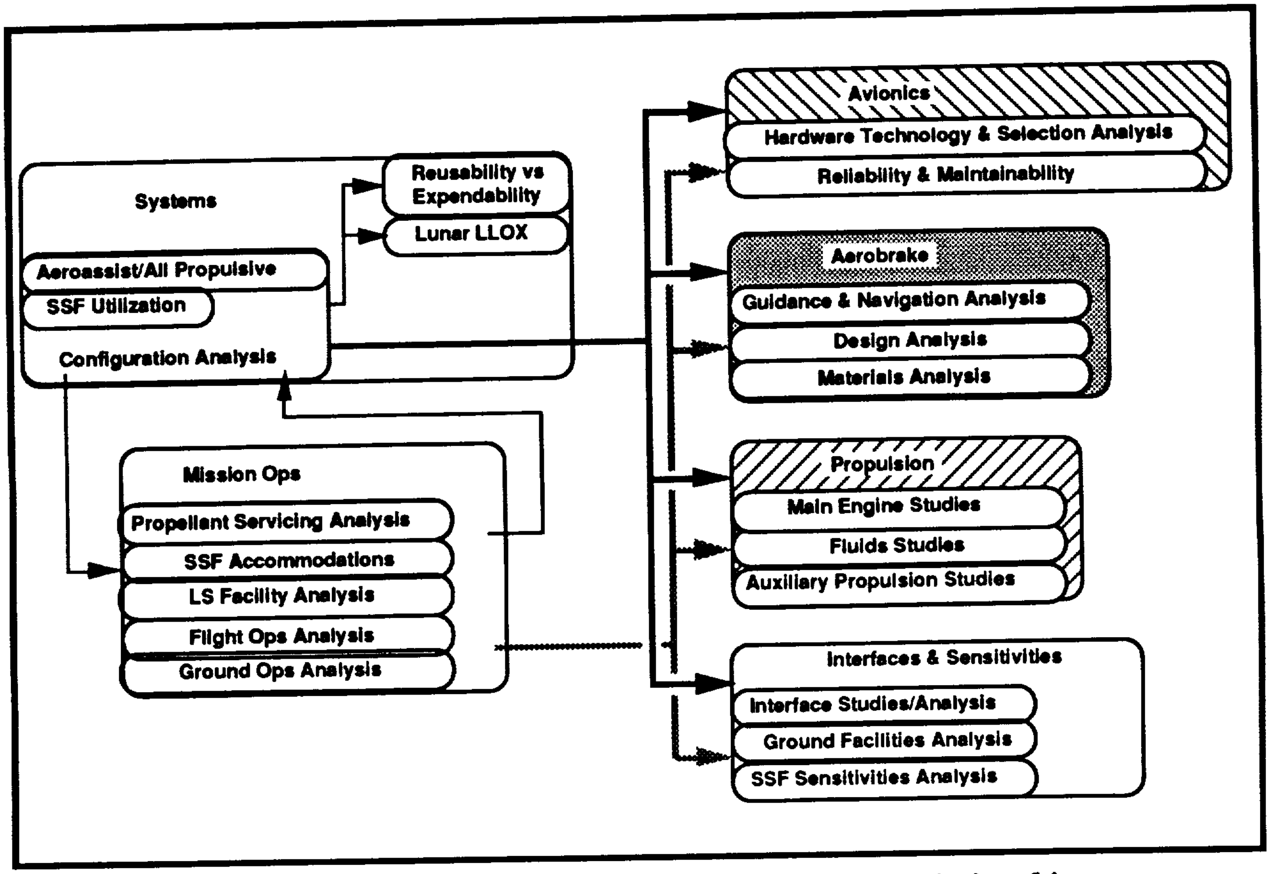


Figure 2.3.4-1: System/Subsystem Study and Analysis Relationship

systems). A third set of requirements are driven by the basing mode proposed for the LTS/STV (space basing, free space basing, or expendable). An assessment of these three sets of requirements provided two primary areas of analysis: 1) reliability and maintenance: and 2) guidance, navigation, and control. It is these two areas that are discussed in detail is the following sections.

2.3.4.1.1 Ground Rules—All mission critical avionics items would be designed with the appropriate degree of functional redundancy (dual, triple, quadruple) as determined by supporting mission and design requirements. Avionics items excluded from this requirement are primary structures, and passive subsystems which have no credible failure modes.

1) Structures. Vehicle structures are excluded from redundant design consideration except for those items listed below:
   a. Structural hardware with movable, pivoting, sliding, expansion, or otherwise flexible joints. (In these cases, the design is robust and contain no credible failure modes.)
   b. Component housings, and attach fittings.
2) Attitude Control and Propulsion Systems. These systems will preclude single point failures. Design considerations would include:

a. Series redundancy precluding loss of RCS propellant.

b. Multiple RCS propellant tanks as feasible.

c. Multiple ignition sources if ignition source required.

d. Multiple pressure relief sources if over pressure is a credible failure mode.

e. At a minimum, multiple engine control valves will exist at the control source.

3) Electrical Power and Cables Systems. The design of these electrical systems will preclude single point failures. Design considerations include:

a. Harness, connectors, and electrical test points shall be independent of redundant systems.

b. Redundant cabling shall be physically separated.

c. Redundant electrical systems shall be independent of common power sources or power returns.

d. Critical commands and data between redundant systems being shared shall be electrically isolated.

e. Redundant electrical power sources shall be electrically isolated.

f. Redundant electrical power sources shall be physically separated.

g. Conditions for two redundant power sources dioded together creating a third source, short circuit protection would be single fault tolerant .

h. Critical electrical heaters required for mission success are redundant and contain series redundant controls.

4) Sensors. The architecture of electrical sensors for critical subsystems are accommodated by single point failure modes. These sensors are the responsibility of the using subsystem, and include requirements for fault detection and fault management.

a. Redundant sensor elements will facilitate early detection of failed operations. (Built-in-test)

b. Redundant sensor elements, either at the sensor element or the in-flight computer processing function, use selection filters for fault detection, and failure mode softening.

c. Redundant critical sensor functions will operate with a failed sensor.

5) Flight Computers and Data Distribution. Single fault tolerance in flight computing systems and the associated data distribution systems, requires a redundancy level of a minimum of three, or the extensive use of dissimilar redundancy. Some of the key elements of conceptual redundant systems design are:

a. Built-in-test and health monitoring is an integral part of normal systems operations.

b. Stand-by, or dissimilar redundant elements are verifiable during all operating modes.

c. Redundant elements will share common components without physical and electrical isolation.

d. Cross strapping of data busses between redundant functions are isolatable so that failure mode propagation is prohibited.

e. The clocking function for digital function is separate and isolatable.

f. Synchronization of critical redundant functions will accommodate single failures and propagation timing delays.

g. The assignment or allocation of common data busses is fixed or controlled by a single-fault tolerant special purpose hardware device.

h. Command channels will not be shared by redundant systems.

6) Redundant Flight Control Actuators. Existing and proposed flight control actuators use various types of redundant architectures. The following designs are complex due to different systems in operation; however all contain key elements of redundant actuator designs.

a For a dual-redundant actuator design using electrical mechanical drive systems, the following design elements should be incorporated:

- The design should insure only one actuator being in control at any single instant of time.
- Built-in-test are continuously conducted in both redundant elements.
- Detection of a failed primary actuator resulting in switch-over and total deactivation of the primary actuator.

b. For a tri-redundant actuator design, the following elements are incorporated:

- Force fight techniques methodology shall be used to soften failure mode until re-configuration is accomplished. Design to accommodate one actuator failed hard-over.
- Bypassing the operation of one leg of a redundant actuator is provided.
- Built-in-test functions are included as a normal operational mode.

7) Redundant Software Elements. Several key software design requirements which must be considered in relation to redundant systems architectures including:

a. Where possible, identical code will be designed for redundant software functions.

b. Cross-strapping of recursive data are prohibited.

c. Data equalization (starting/re-starting) is not used unless at a quiescent point and verifiable by external systems.

d. Minimum data interchange between redundant function is a design goal.

e. Software process control will be fixed by design or methods of process synchronization should be part of the systems design.

8) Ground Support Equipment (GSE). Some design requirements for single fault tolerant systems spill over even to the ground support equipment used in test or assembly of hardware. The following are guidelines for GSE tasks:

a) Redundant functions will not share common GSE test points.
b) Redundant software function will be loaded, and verifiable by GSE.
c) All redundant functions will be verifiable by GSE.
d) The GSE design will prohibit GSE failure modes from causing loss of redundancy.

2.3.4.1.2 Reliability & Maintenance—The requirement for an avionics design to incorporate redundant systems is a reliability issue. In the past, the level of redundancy for a specific program has been resolved through a quantitative risk assessment effort during the conceptual design phase. Currently, NASA sponsored space programs are requiring redundancy for mission and safety critical subsystems independent of these traditional reliability calculations, therefore for manned rated systems, the level of redundancy has been set at two fault tolerant for man-safe designs, and three fault tolerant for man-rated systems.

2.3.4.1.2.1 Reusability—Two missions have been derived from the requirement for reusability; the expendable system and a system which requires periodic servicing. The requirements derived from reusability are a direct function of mission design. Manned safety aspects are derived by the requirements defined in section 2.3.4.1.1. Achieving a reliability above 96 percent for electrical devices requires the use of redundant 'black boxes'. Electronic subsystem designs for space vehicles have achieved outstanding reliability results, although the cost to maintain them is considerable. For example, during the design phase of the shuttle program long term maintenance requirements received little emphasis. The recommendation from the STATS conference was that future systems maintainability should be addressed beginning in Phase A in order to achieve the required reliability levels.

2.3.4.1.3 Navigation, Guidance, and Control—Two types of navigation systems are required for deep space missions, 1) short term navigation using an inertial navigation unit, and 2) and navigation updates using either ground-based ranging or on-board autonomous navigators. Numerous inertial navigation units are available which provide short term inertial navigators. A discussion of the characteristics of these systems is provided in this section. As a reference in the assessment of potential systems, NASA's long range inertial navigation up-date system is the pseudo-random ranging system built during the 1960's for both manned and unmanned space operations, and incorporated into the Deep Space Network (DSN). Although adequate for unmanned space operations it does not meet the fault tolerance requirements of a manned mission without some form of augmentation.

2.3.4.1.3.1 Short Duration Navigation—The Transfer Orbit Stage (TOS) program uses a short duration navigation system, known as the LINS (Laser Inertial Navigation System). This system represents the most modern, qualified navigation system available for use in space vehicles. A second generation of laser navigation systems is presently under development for the Titan IV/Centaur program, while on the horizon a new set of inertial sensors configured in a hex-head configuration are being pursued. It is this last set of inertial sensors which is of direct interest for use in the LTS/STV systems, since the potential exists for lower power, weight, and volume in inertial navigation systems.

2.3.4.1.3.2 Long Duration Navigation—Several disadvantages appear to be paramount in the use of the Deep Space Communication Network (DSCN) for long duration navigation. Power, weight and volume of S-band or KU-band communication systems require large steerable antennas which must be pointed in the direction of the ground stations line-of-sight; also the transponders are insufficient due to weight and considerable power usage. The complexity of these communication systems to meet the FO/FO/FS requirements of a manned systems are substantial.

On-board optical navigators, represent a new approach for long duration autonomous navigation. Several potential optical systems are available for this function. These new generation systems provide the greatest promise for lowering the weight, power and volume of navigational 'black boxes'. The applications of these optical systems to space operations appears straight forward. Figure 2.3.4.1.3.2-1 graphically defines the landmark navigation approach using line-of-sight optical systems. One aspect of this application is the possibility of multiple uses. Navigation and attitude alignments, as well as automated docking and lunar landing can be accomplished. This approach substitutes automated systems that in the past have been manned functions. From a mission design standpoint, one mission scenario for manned and unmanned vehicle operations is achievable.

Rendezvous navigation can use a similar approach to the problem of navigating on either the space station or the LTS/STV. The mathematics of the problem evolve around the ability of navigation sensors to determine the line-of-sight between the two orbiting bodies. Optical sensors with the ability to discriminate the target from background clutter have the potential to provide relative navigation parameters needed for rendezvous navigation. Figure 2.3.4.1.3.2-2 shows an optical navigation sensor providing line-of-sight navigation data needed for rendezvous.

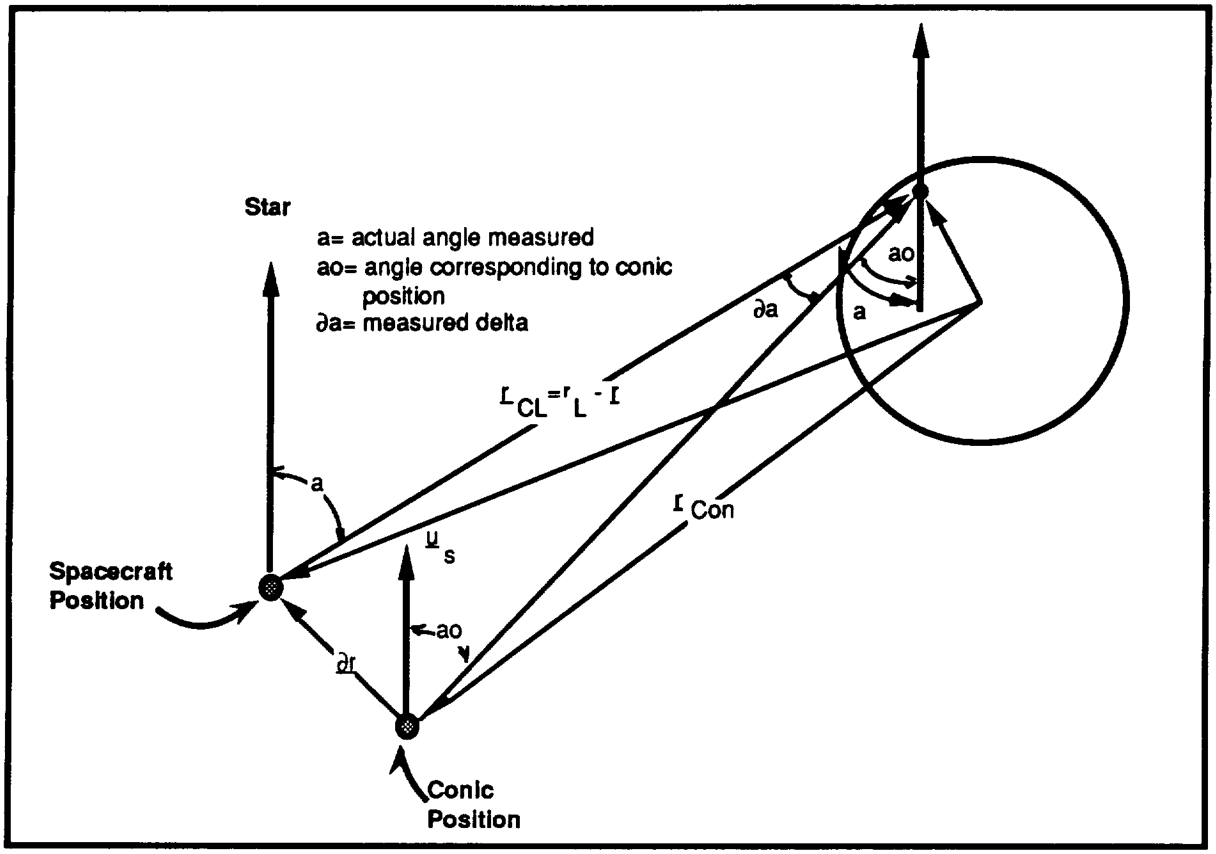


Figure 2.3.4.1.3.2-1 Landmark Navigation Approach

2.3.4.1.3.3 Guidance —The guidance requirements for the LTS/STV missions are similar to those employed during the Apollo missions with the exception of the aerobrake deceleration system. The Lambert guidance scheme, currently implemented in NASA's manned space systems adequately meets these requirements for conducting lunar missions.

Martin Marietta proposes that the LTS/STV program baseline Lambert guidance for long duration main propulsion maneuvers, cross product guidance for short duration maneuvers, and explicit guidance for lunar landing. Since all of these guidance schemes have been man-rated in the past, there exists considerable documentation within NASA on each of this guidance system which can be directly applicable LTS/STV program.

2.3.4.1.3.4 Flight Control Systems—The present LTS/STV configuration requires a large liquid propulsion system which needs to develop a robust control system for main propulsion maneuvers. The concept of robust control systems is supported throughout the aerospace avionics community. Technology emphasis appears promising for advanced systems like the LTS/STV which will have time to develop the concept further. This type of robust control system provides the ability to verify and validate the final control system. Previous phase/gain approaches used on launch vehicles in the past have proven costly to maintain and verify for each new mission and flight configuration to which the vehicle is subjected. The LTS/STV should baseline this type of robust control system even though further development over the next few years will need to be accomplished.

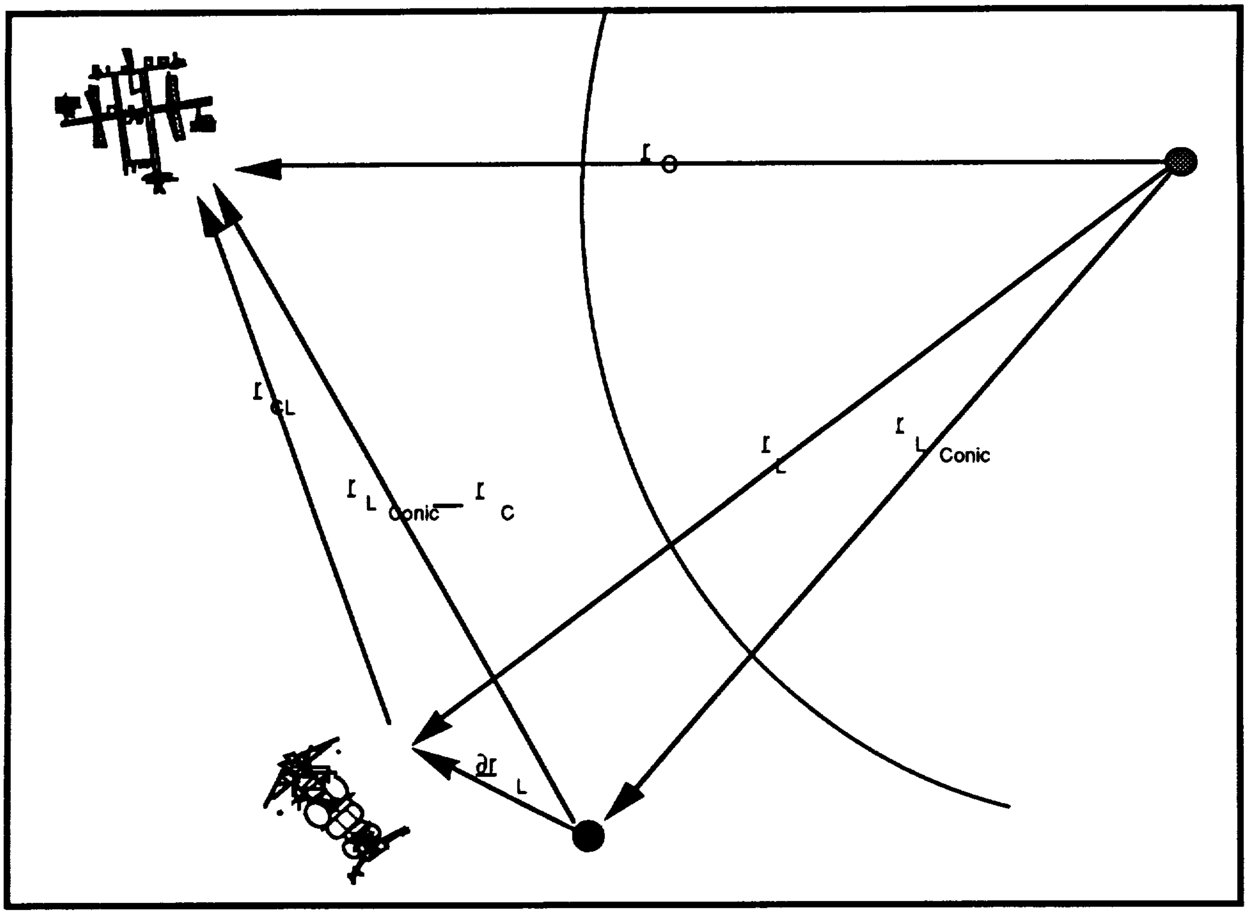


Figure 2.3.4.1.3.2-2 Optical Navigation As Used for Rendezvous

RCS attitude control of the LTS/STV flight vehicle is primarily driven by the response control authority of rendezvous maneuvers and the aerobrake roll control authority. The sizing and location of the RCS thrusters will be discussed in detail as part of the propulsion analysis in section 2.3.4.2.

2.3.4.1.3.5 Telemetry, Command and Tracking—The telemetry, command and tracking requirements for the LTS/STV are defined by the ground station capabilities of the TDRS and Deep Space Network. Presently these systems are S-Band transponders supporting encoded command data rates of 128 bits per second and transmission data rates of 64 and 128 kb/s. Significant technology advancements are not required to support this function other than a desire to lower the weight, power and volume of existing systems. This function is assumed to be relegated to a mission success requirement, and therefore, a single fault tolerant design similar to the shuttle should be baselined.

2.3.4.1.3.6 Timing and Sequencing—To meet the autonomous navigation requirements of the LTS/STV missions, an accurate knowledge accounting of time is required. Since up-dates from the ground are assumed to be mission critical, an atomic clock, which can be temperature controlled, is the primary candidate for extensive development activities. The long term drift of the timing system must meet the autonomous mission duration times. Mission sequencing of flight activities derived from the on-board timing system are a normal requirement for manned and unmanned space vehicles. The overall LTS/STV mission although not complex, requires scoping of the interaction of subsystems operation, health monitoring, and mission sequencing as the system matures.

2.3.4.1.3.7 Mechanical Subsystem Controls—The present LTS/STV concept contains several different types of mechanical systems such as tank separation subsystems, deployment subsystems, and docking subsystems. The controls for these subsystems are not yet completely defined, however the requirement for the task is part of the total systems decomposition. The redundancy requirement for the controls of these systems will be consistent with flight safety requirements; jettisoning of the structure to assure flight safety may become a requirement.

2.3.4.1.3.8 Electrical Power & Control—The generation of electrical power for the LTS/STV remains undefined at this point, due to the maturity level of the systems and subsystems that require it. Regardless of the amount of power required, the redundancy level of the system will meet the FO/FO/FS requirement.

2.3.4.2 Propulsion Analysis

2.3.4.2.1 General Parametrics

2.3.4.2.1.1 Performance of Potential Engine Candidates

Although cryogenic propellant was the primary baseline for the STV study, three types of engines were initially evaluated as candidates for use on an LTV and LEV vehicle to become familiar with some of the system performance parametrics associated with different engine types. These engine types were cryogenic, storable, and Nuclear Thermal Rocket (NTR). Cryogenic engines fall into essentially two categories - Advanced Space Engine (ASE) and RL-10 derivatives. The storable engines also fall into two categories - pump-fed, such as the XLR-132 engine and pressure-fed, most likely the Apollo Lunar Excursion Module (LEM) descent engine. The NTR engine would be based on NERVA (Nuclear engine for Rocket Vehicle Applications) technology developed in the 1960's. A listing of these engines and their characteristics is shown in Table 2.3.4.2.1.1-1.

Table 2.3.4.2.1.1-1 Potential Engine Candidates

Engine Type	Application	Isp (sec)	Thrust* (N)	Mass (kg)
ASE	LTV	481	88960	220
ASE	LEV	465	86180	213
RL-10B-2	LTV	460	97856	206
RL-10B-2	LEV	446	94878	200
RL-10A-4	LTV & LEV	449	92518	166
XLR-132	LEV	347	88960	168
LEM Descent Engine	LEV	305	88960	300
NTR	LTV	900	133440**	5443

* Thrust per Engine
** NTR: Only 1 Engine Was Used,
In All Other Cases 4 Engines Were Used

A combined LTV/LEV sizing and performance model was used to compare the total effect of using different engine combinations on the two vehicles. The "90 day configuration" was used for this study. The weight of the cargo and the modules was held constant, and the IMLEO required to perform the mission was calculated for each of the engine and vehicle combinations. The payload capability of the Shuttle-C is 71 tonnes with the 15 foot diameter shroud and 61 tonnes with the 25 foot diameter shroud. Thus the cryogenic engine configurations require 3 Shuttle-C flights, most of the configurations which use storable engines for the LEV require 4 flights, and two of the storable configurations require 5 flights. The nuclear engine LTV configuration shows a clear advantage, requiring only two Shuttle-C flights. These relationships are shown in Figure 2.3.4.2.1.1-1.

If the data from Figure 2.3.4.2.1.1-1 are plotted relative to the baseline ASE performance for both the LTV and LEV, the differences are emphasized. This is shown in Figure 2.3.4.2.1-2. Using RL-10 derivative engines incurs a modest weight penalty (<10%), but using storable engines for the LEV incurs a much larger penalty, particularly if the pressure-fed, lower Isp lunar Excursion

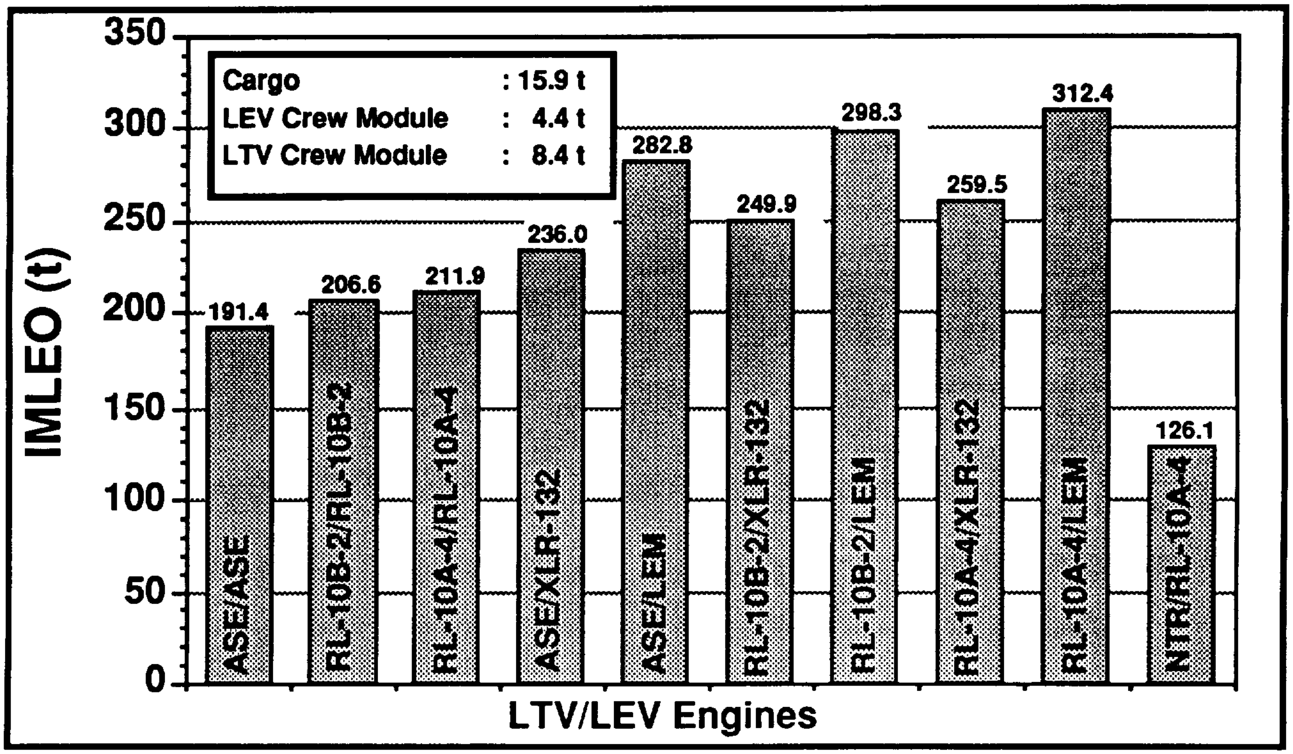


Figure 2.3.4.2.1.1-1 IMLEO vs Engine Selection For First Flight

Module engine is used. The reason for using the storable propellant would be to avoid the problem of cryogenic propellant boiloff issues during the 30 day lunar stay of the astronauts. Use of a NERVA type nuclear engine for the LTV provides a weight saving of over one third (34%) compared to the ASE baseline.

2.3.4.2.1.2 General Propulsion Parametrics

General propulsion parametrics were developed around the performance of the "90 day configuration". An analysis was conducted to determine the effect of different numbers and thrust levels of engines for the LTV. The results are shown in Figure 2.3.4.2.1.2-1. The LTV propulsion is analyzed because it operates near earth, where gravity losses play a significant role in the efficiency of a vehicle stage. Once the vehicle is in the lunar vicinity, gravity losses become much less significant. If an LTV has only one or two engines, the resulting thrust-to-weight ratio is very low and the vehicle incurs large gravity losses. As the number of engines increases, the gravity losses quickly approach zero, but at some point the weight of the engines overpowers the reduction in gravity losses, and the IMLEO begins to increase. This crossover point occurs at 5 engines for the larger 30,000 lbf engine. This curve also indicates that a total thrust of 60,000 to 90,000 pounds is desirable when using these engines, to avoid a large increase in gravity losses.

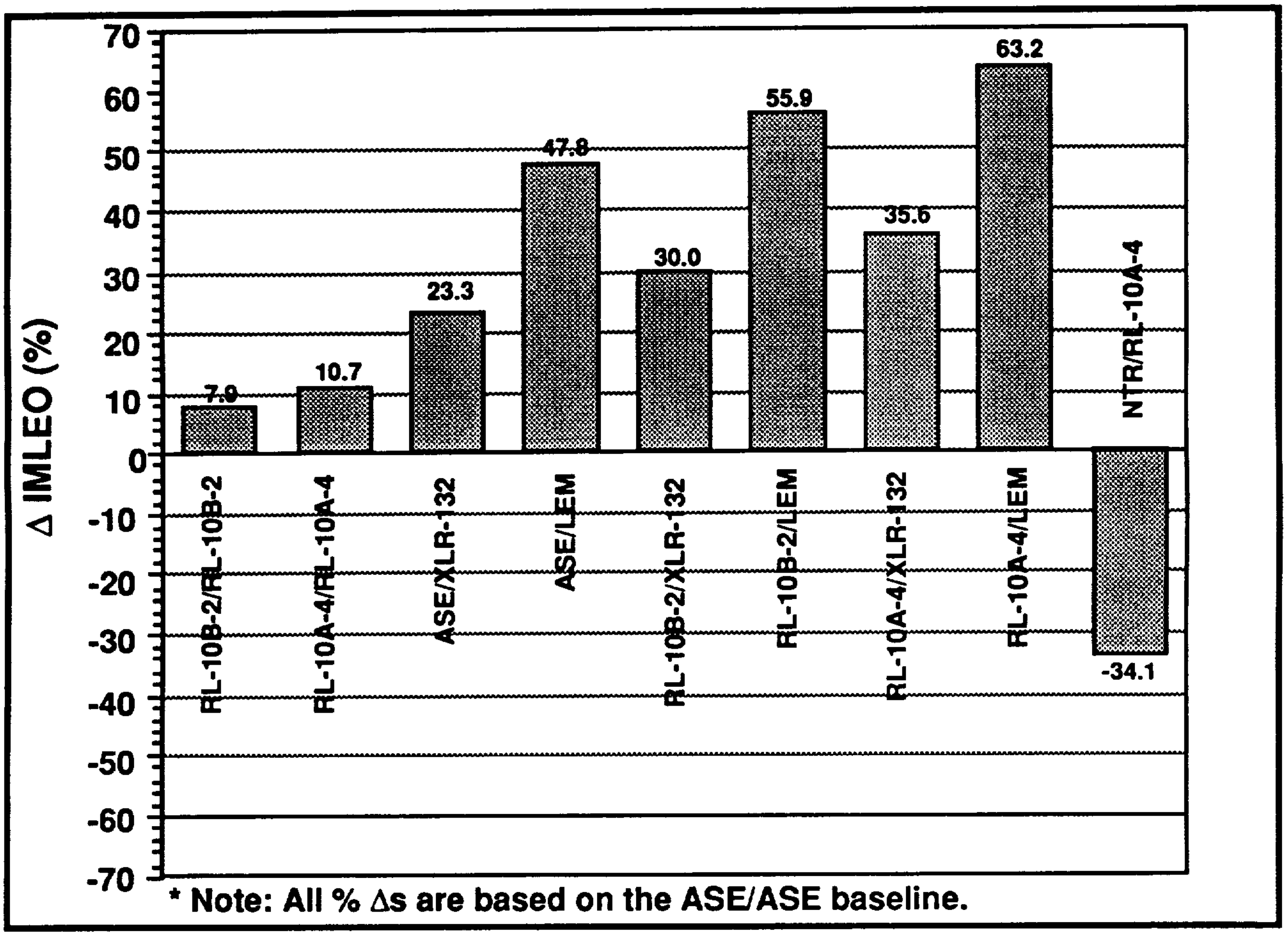


Figure 2.3.4.2.1.1-2 Relative Change in IMLEO vs Engines

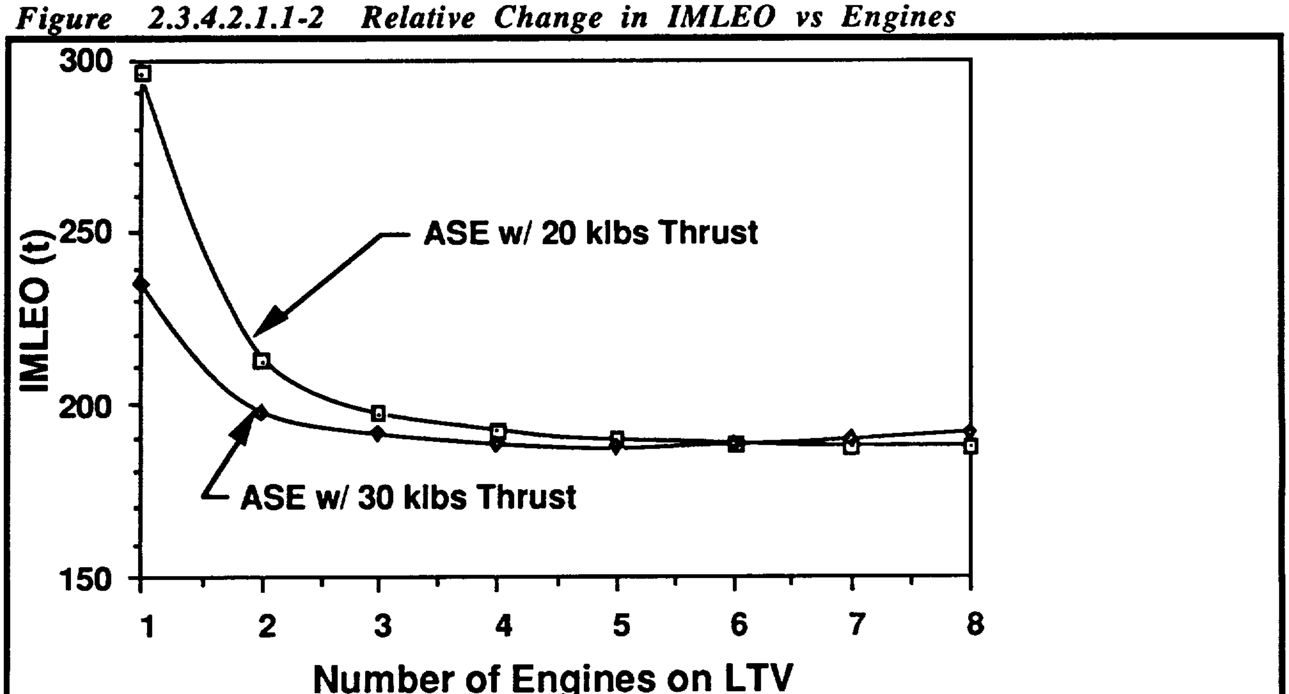


Figure 2.3.4.2.1.2-1 IMLEO vs Number of LTV Engines

For the charts shown in this section, it is assumed that the LEV has 4 ASE engines of 20,000 lbf thrust each. It is also assumed that this is the first flight of a LTV, which will include the LEV and cargo. It is assumed that the lunar cargo delivered is 15.9 tonnes, that the LTV crew module weighs 8.4 tonnes, and that the LEV crew module weighs 4.4 tonnes. Isp effects were not included in the calculation of gravity losses.

Figure 2.3.4.2.1.2-2 shows the relationship between thrust-to-weight, IMLEO, and Isp of an LTV. Over the range of values considered for Isp and thrust-to-weight, the IMLEO may be seen to vary nearly 2:1. The Isp ranges from 340 sec, which corresponds to pump-fed storable engine

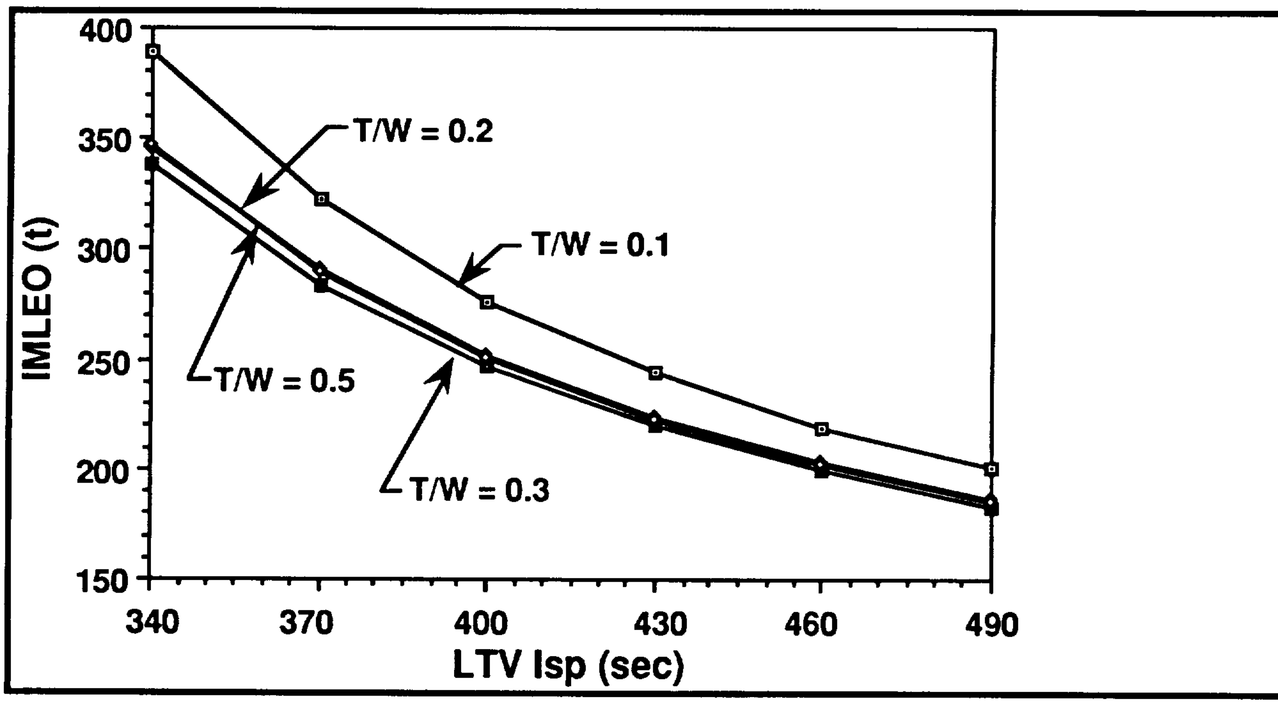


Figure 2.3.4.2.1.2-2 IMLEO vs LTV Isp

performance, to 490 sec, which is slightly above the highest possible cryogenic engine performance. For RL-10 engine derivatives, the Isp would range from 450 sec to about 465-470 sec, and for an ASE engine the Isp would range from 565 to 481 sec. These curves are plotted for 4 thrust-to-weight ratios. The T/W=0.5 curve lies virtually on top of the T/W=0.2 curve and above the T/W=0.3 curve because the increased engine weight has overpowered the gravity loss reductions. For this chart, it is assumed that the thrust-to-weight of the LTV engines is always 40:1.

Figure 2.3.4.2.1.2-3 directly illustrates the effect of thrust-to-weight on the amount of mass required in earth orbit. For a given thrust-to-weight ratio, the amount of gravity losses can be reduced slightly by breaking the tans-lunar injection burn into two burns.

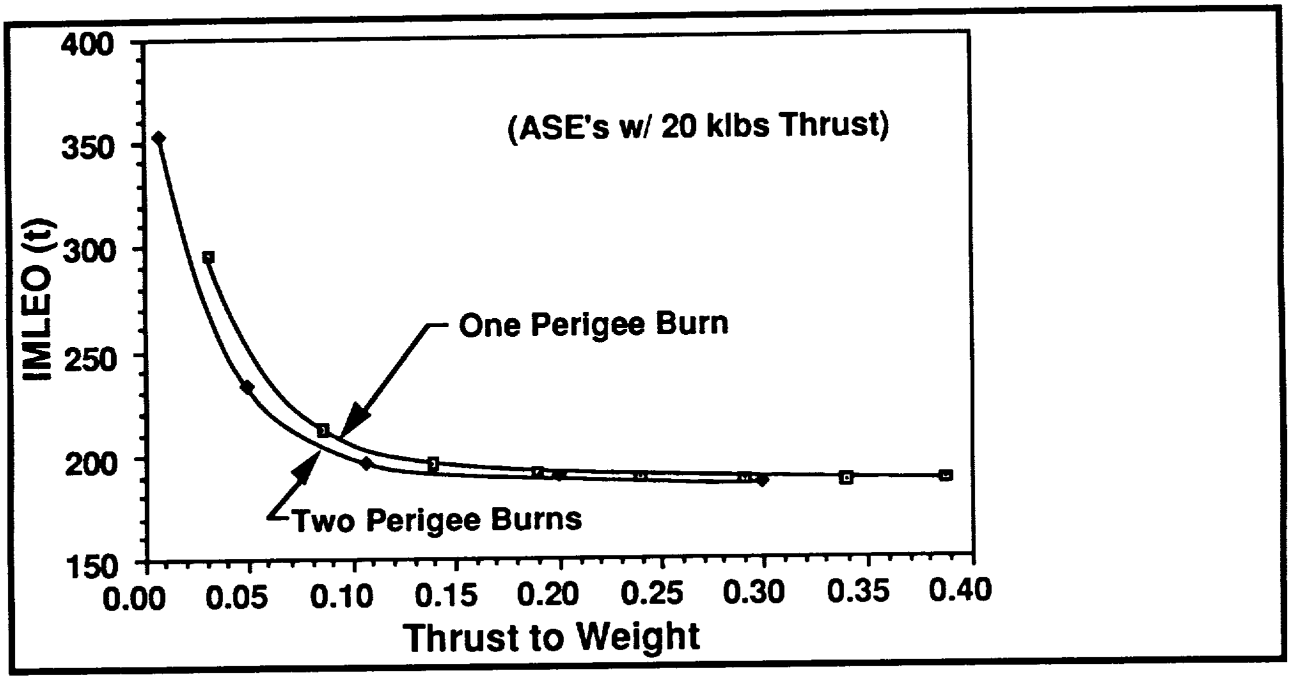


Figure 2.3.4.2.1.2-3 Effect of Dual TLI Perigee Burns

2.3.4.2.2 General Engine Data

2.3.4.2.2.1 Engine Criteria Evolution

Engine characteristics required for a steady state piloted lunar vehicle are more advanced than those required for other DRM's or precursor missions. It is appropriate to let the engine criteria evolve to meet the mission requirements without requiring all the resource investment to be made at the outset. Figure 2.3.4.2.2.1-1 connects the simpler criteria associated with the beginning of the evolution path as meeting initial missions, and connects the more advanced of the evolved criteria as appropriate to steady state piloted lunar missions. This is intended to represent more of a philosophy than a mature understanding of the subject, as there are many more criteria and many more gradations than shown here. It is hoped that others may expand upon this technique to help prioritize the resource expended in the propulsion areas.

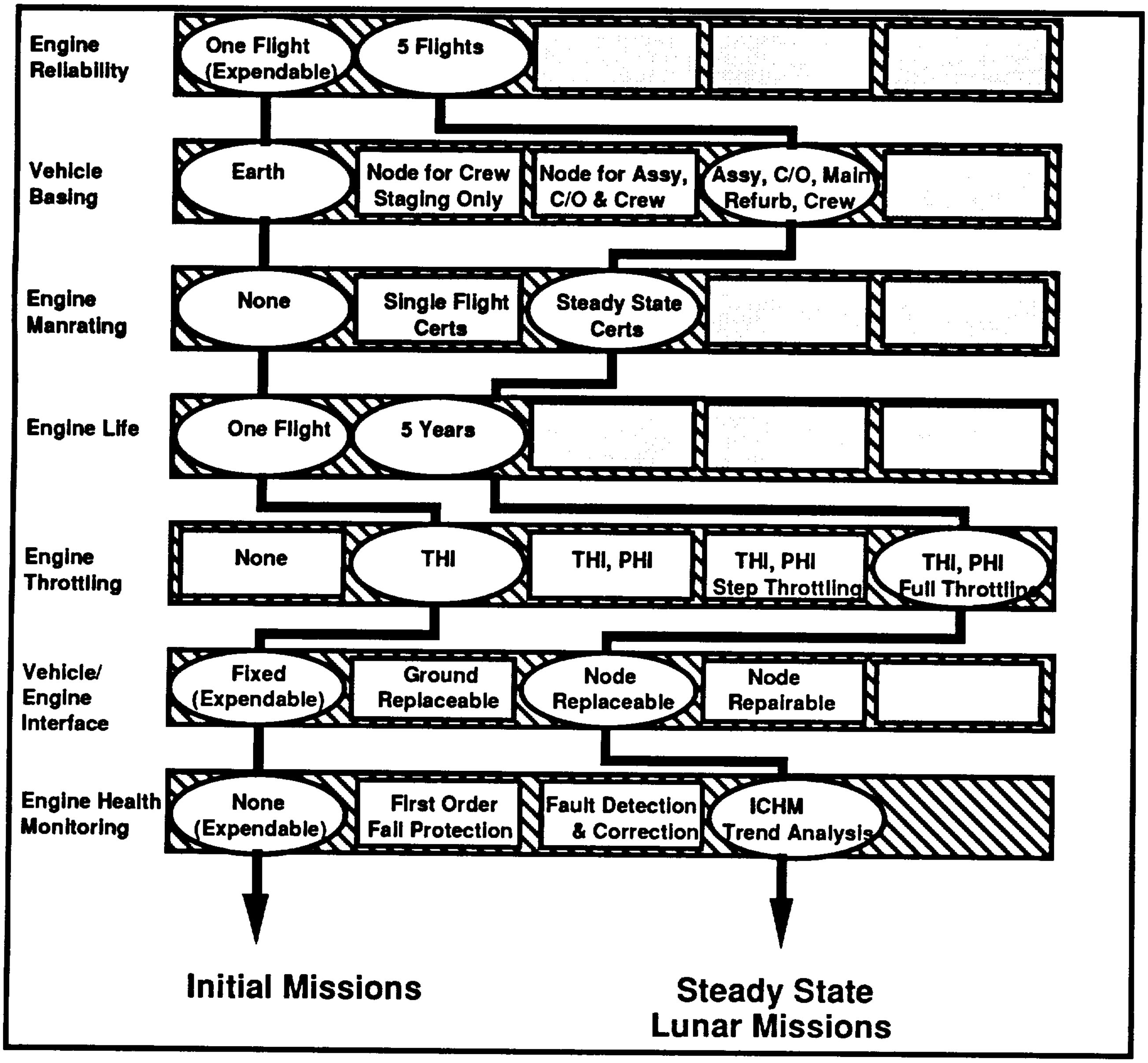


Figure 2.3.4.2.2.1-1 Engine Criteria Evolution

2.3.4.2.2.2 STV Main Engine Change-out Strategy

Using a single robotic arm equipped with engine handling fixture, and an engine assembly equipped with a pneumatically actuated carrier plate for grouping of individual disconnects, removal and replacement of an LTV main engine may become a relatively normal maintenance task. Figure 2.3.4.2.2.2-1 is a representation of a potential engine changeout scenario. The plan would be to change the entire engine assembly, rather than individual parts of an engine, such as pumps, valves, etc. A robotic arm is shown with a probe that would fit into the engine combustion chamber for support and movement of the engine. The probe would be pre-programmed for replacement of each of the engines. The scenario shown may be realistic if the engine installation is planned from the outset to include disconnect assemblies as a part of a carrier plate for ease of replacement.

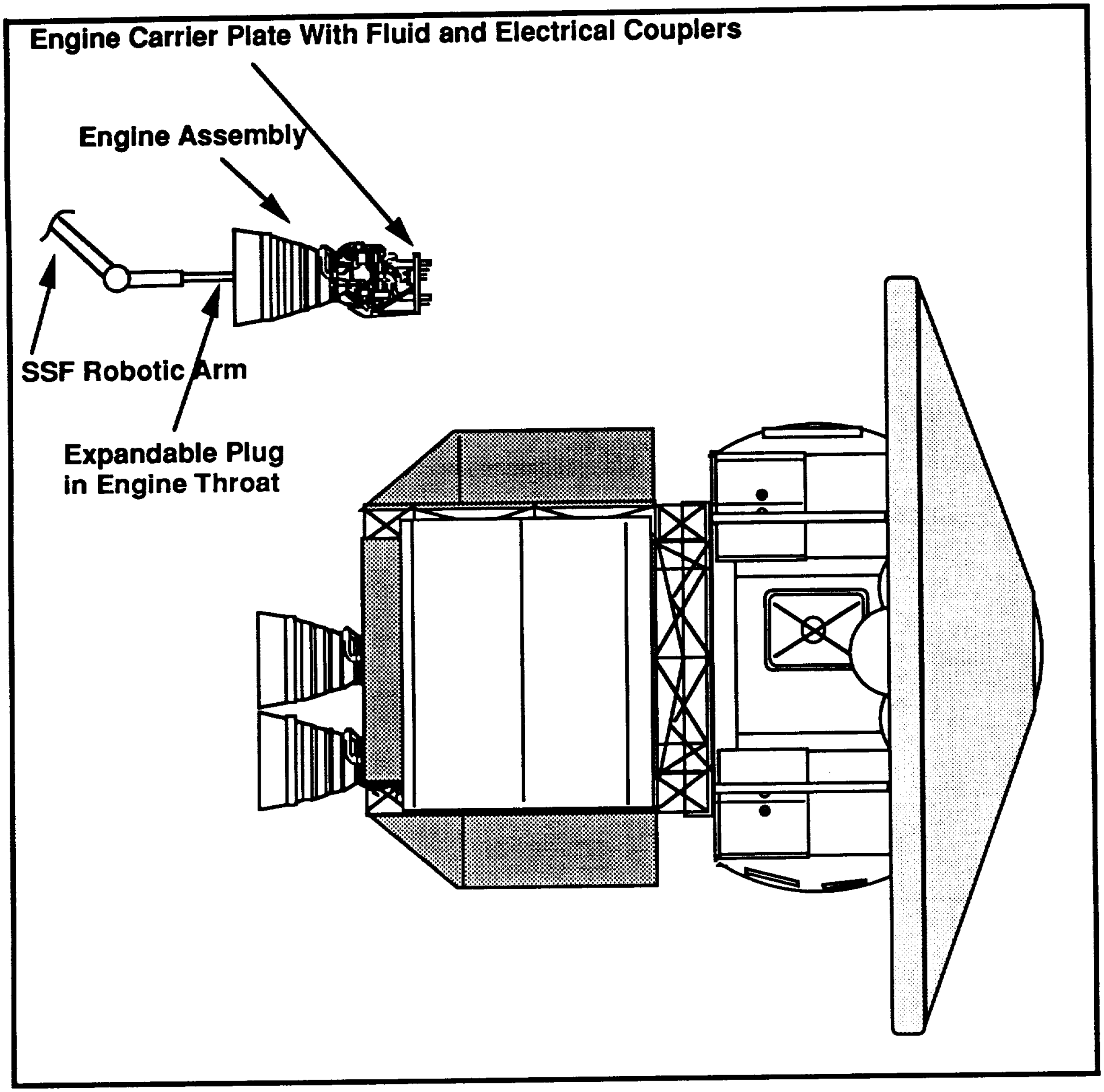


Figure 2.3.4.2.2.2-1 STV Main Engine Change-Out

Figure 2.3.4.2.2.2-2 is a more detailed view of an arrangement that could be used for engine mounting. It shows a vehicle carrier plate that is incorporated into the lower portion of the box beam engine support. It also shows that the engine is first assembled onto an engine carrier plate, that incorporates all the engine interfaces, and which mates with the vehicle carrier plate disconnects.

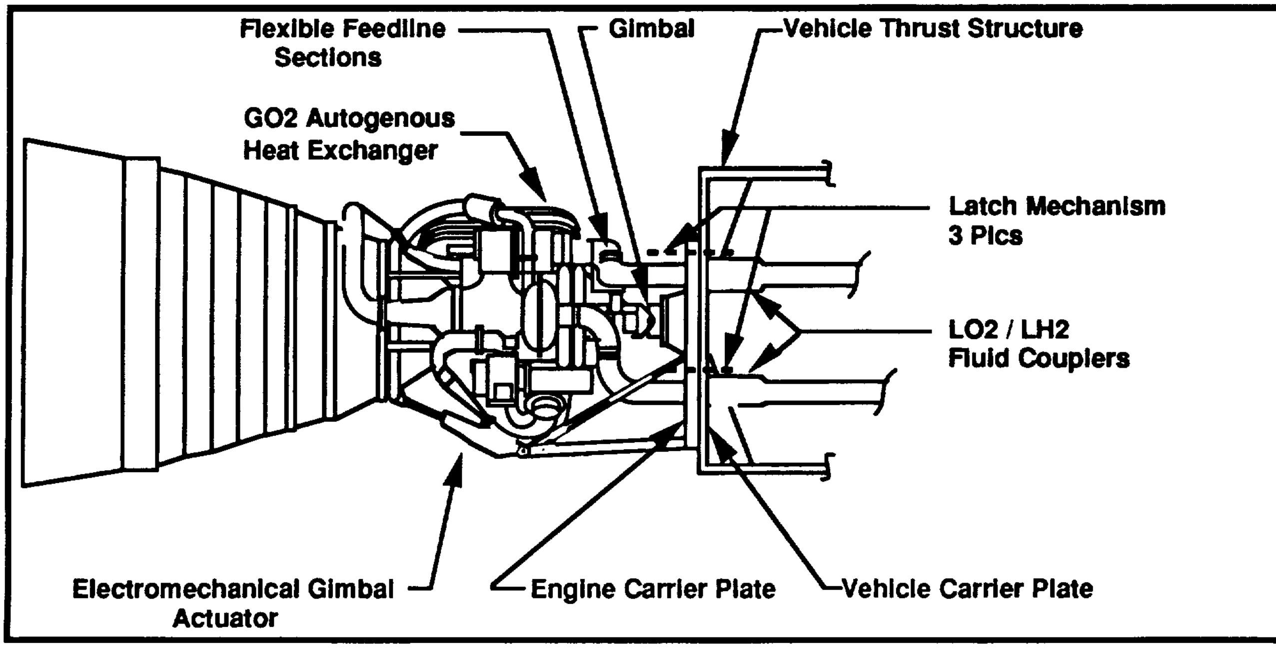


Figure 2.3.4.2.2.2-2 Engine Replacement Architecture

Additional detail of a potential layout for an engine carrier plate is shown in Figure 2.3.4.2.2.2-3. All the equipment in this illustration would be a part of the engine assembly. The disconnects shown would all penetrate the vehicle carrier plate and be locked into place to install the engine. Engines to be developed would use a common interface arrangement, that would allow different engine versions to be installed for upgrades or for tailoring to a specific mission.

2.3.4.2.2.3 Engine Replacement Times

To assess the practicality of a timely replacement of an STV engine, a tabulation was prepared of accepted engine replacement times for launch vehicles and other "high-tech" engines. This appears in Table 2.3.4.2.2.3-1. The hours are broken into two categories, the time to remove and replace, and the time to perform the various quality and checkout steps to get to the point of "run-up". For comparison to the time required for an STV, which should have automated checkout capabilities, through the ICHM system, only the remove and replace time is considered applicable. The time required for these ground operations is approximately one to two shifts.

Figure 2.3.4.2.2.3-1 presents an estimated timeline for robotic engine removal and replacement, assuming a carrier plate system has been designed into the vehicle for engine attachment. It lists each of the activities to be accomplished, assumes that the robotic sequencing is mature, and that

Table 2.3.4.2.2.3-1 Engine Replacement Times

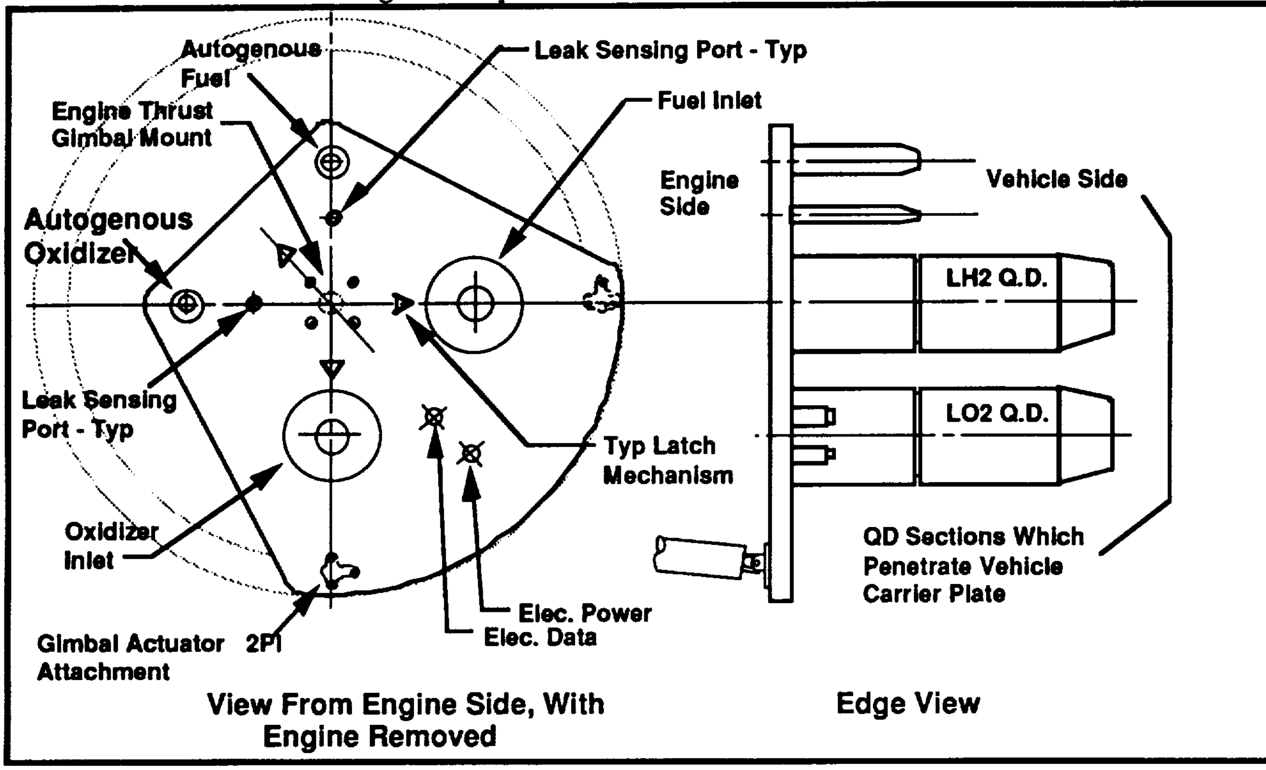


Figure 2.3.4.2.2.2-3 STV Main Engine Carrier Plate

Table 2.3.4.2.2.3-1 Engine Replacement Times

Vehicle	Engine	Span Hours Remove & Replace	Span Hours QC & Checkout	Span Hours Total
737	P&W J78D-9	10	Included	10
727	P&W J78D-9	11	Included	11
MD-80	P&W J78D-9	10	Included	10
Shuttle	SSME	16	16	32
T-IV (1st Stg)	AJ LR-87	16	64	80
T-IV (2nd Stg)	AJ LR-91	8	48	56
Delta (1st Stg)	MB3	N/A Boat Tail Config Requires Return to Factory		
Delta (2nd Stg)	AJ 10	24	80	104
Centaur	RL-10	16	32	48
Atlas	MA 5	N/A Requires Return to Factory		
Saturn	H1	32	Included	32
Cobra Helicopter	Bell	8	Included	8
TOW Missile Carrier	Detroit Diesel	4	Included	4
Chevrolet	454 V-8	7.5	Included	7.5

the necessary tools and prepared replacement engines are readily available. The initial attempt at an operation of this magnitude would likely require significantly more time.

2.3.4.2.3 Number of Engines

There are a number of issues associated with selecting the number of engines for a LTS vehicle. The following sections explore the requirements and issues, and conclude that the required number for a landing vehicle that desires centerline thrust following engine failure(s) is 5 engines, and that the required number for a transfer vehicle that has engine gimbals is 4 engines.

2.4.3.2.3.1 Number of Engines for a Lander

Table 2.3.4.2.3.1-1 examines the issues related to different numbers of engines for a lunar landing vehicle. Both positive and negative issues are considered. A key issue is the ability of the propulsion system to tolerate failures. Configurations with one or two engines cannot tolerate any failures and still maintain symmetrical thrust for a level landing attitude. A configuration with three engines in a line can tolerate a single failure. This is a loss of an outer engine requiring shutdown of the opposite engine, or loss of the inner engine leaving two outside engines. A four-engine configuration can also tolerate one engine failure - it also requires an opposite engine to be shut down to maintain symmetrical thrust. A five- engine configuration, with one engine in the center, can tolerate two failures. One more level of failure tolerance (3 failures) would require a seven-engine configuration, with six engines arranged around a center engine. It is interesting that the MASE requirement quoted at the bottom of the table makes the safe return of the crew and the "Program Elements" essentially equal.

2.3.4.2.3.2 Landing Control

MASE document Human Exploration Study Requirements, March 14, 1990, requires that "Critical functions affecting crew safety and Human Exploration Program Element (HEPE) survival shall be two-failure tolerant." If this philosophy is applied to engines, the minimum number of engines that will meet this requirement is four (4). If an additional requirement for maintaining centerline thrust after engine failure(s) during lunar descent or ascent is imposed, the minimum number of engines required increases to five (5). There are two items that prompt the derived requirement for centerline thrust. The first item is the situation encountered when landing a vehicle with four engines, where both downhill (relative to a slope to be encountered at the landing site) engines

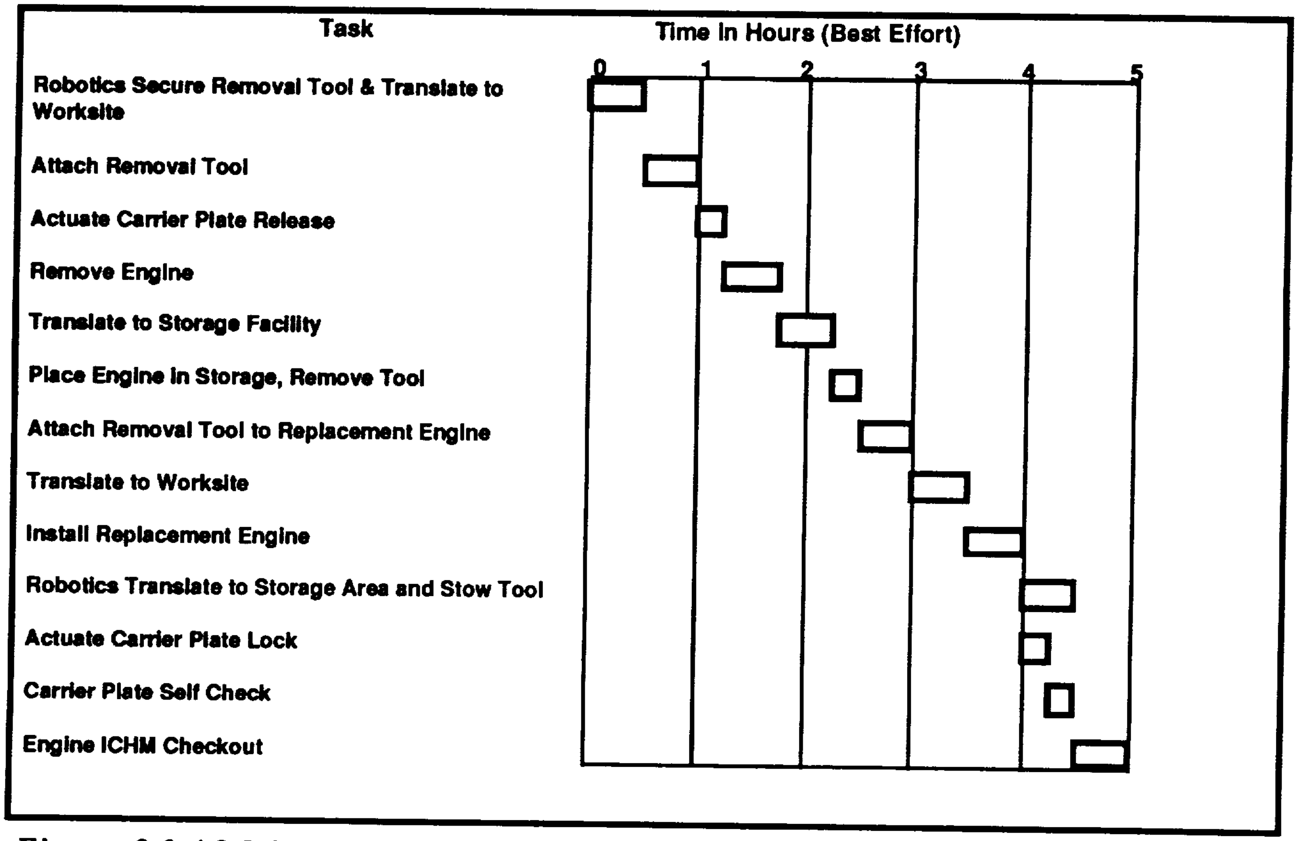


Figure 2.3.4.2.2.3-1 STV/LTV Main Engine Remove/Replace Timeline

Table 2.3.4.2.3.1-1 Number of Engines for a Lunar Lander

Number of Engines	Positive	Negative
1	Symmetrical Thrust	No Engine Out Capability
2		Asymmetrical Thrust After Engine Out
3	Symmetrical Thrust After Engine Out	Provides Only One Level Of Redundancy
4	Symmetrical Thrust After Engine Out Maintains Roll Control	Provides Only One Level Of Redundancy
5 or 6	Minimum No. To Meet Two Failure Tolerance *	

* Reference: Human Exploration Study Requirements, March 14, 1990:
"3.1.7.2 Failure Tolerance
- Critical functions affecting crew safety and Human Exploration Program Element (HEPE) survival shall be two failure tolerant."

have failed, the uphill engines are operating, and are gimbaled as necessary to place the thrust vector through the center of gravity. The vehicle will impact its lowest legs first, and will need to do some carefully timed gimballing and throttling to prevent slapdown or skidding. The ascent from such a scenario would be similarly difficult. This is illustrated in Figure 2.3.4.2.3.2-1. The situation could be improved significantly if sufficient time is available after engine failure to rotate the vehicle 180 degrees about its vertical axis, but if failures occur close to touchdown, or just as the vehicle is leaving the surface, it may be difficult to take the proper corrective action. This leads to the second item that drives this requirement, one of avionics workload. There will likely be a number of pre-programmed reconfiguration scenarios to cover all manner of engine difficulties. If engine gimbal control is used to maintain vehicle orientation and vertical rates during the last few seconds before touchdown, or the initial seconds of ascent, the avionics workload will be significant during this time, and will be category 1 criticality. If five engines are used, the logic becomes simpler and the response times likely not as sensitive.

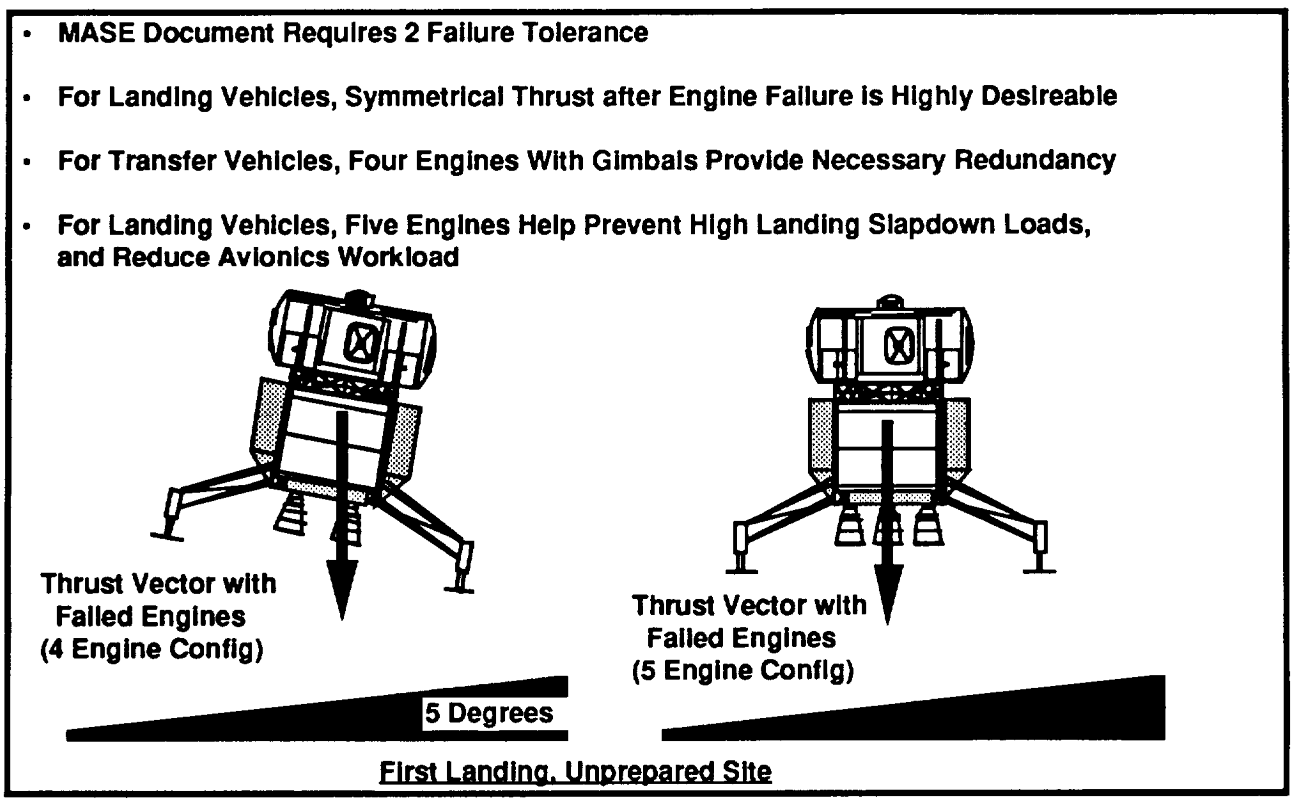


Figure 2.3.4.2.3.2-1 Five Engines Improve Landing Control

2.3.4.2.3.3 Engine Out Strategy

The following three figures explore the logic associated with engine failures, and the reconfiguration of the remaining engines to maintain vehicle control and mission completion. First, a system consisting of four engines is considered in Figure 2.3.4.2.3.3-1, which has been set up to operate two engines at a time, with the other two in idle mode. As can be seen, even with two failures, reconfigurations may be successfully made, but not in a way that would satisfy the centerline thrust requirement we have established for a landing vehicle. A scenario is also investigated that starts with all four engines operating. The outcome is the same, although the options are reduced.

Similarly, a system using five engines is shown in Figure 2.3.4.2.3.3-2, which is intended to have two engines operate normally, with the others in idle mode. In this scenario, the reconfiguration after two independent engine failures always results in a single engine operating in the center, which is quite acceptable for landing or ascent. This configuration maintains a balanced number of engines operating about the centerline always.

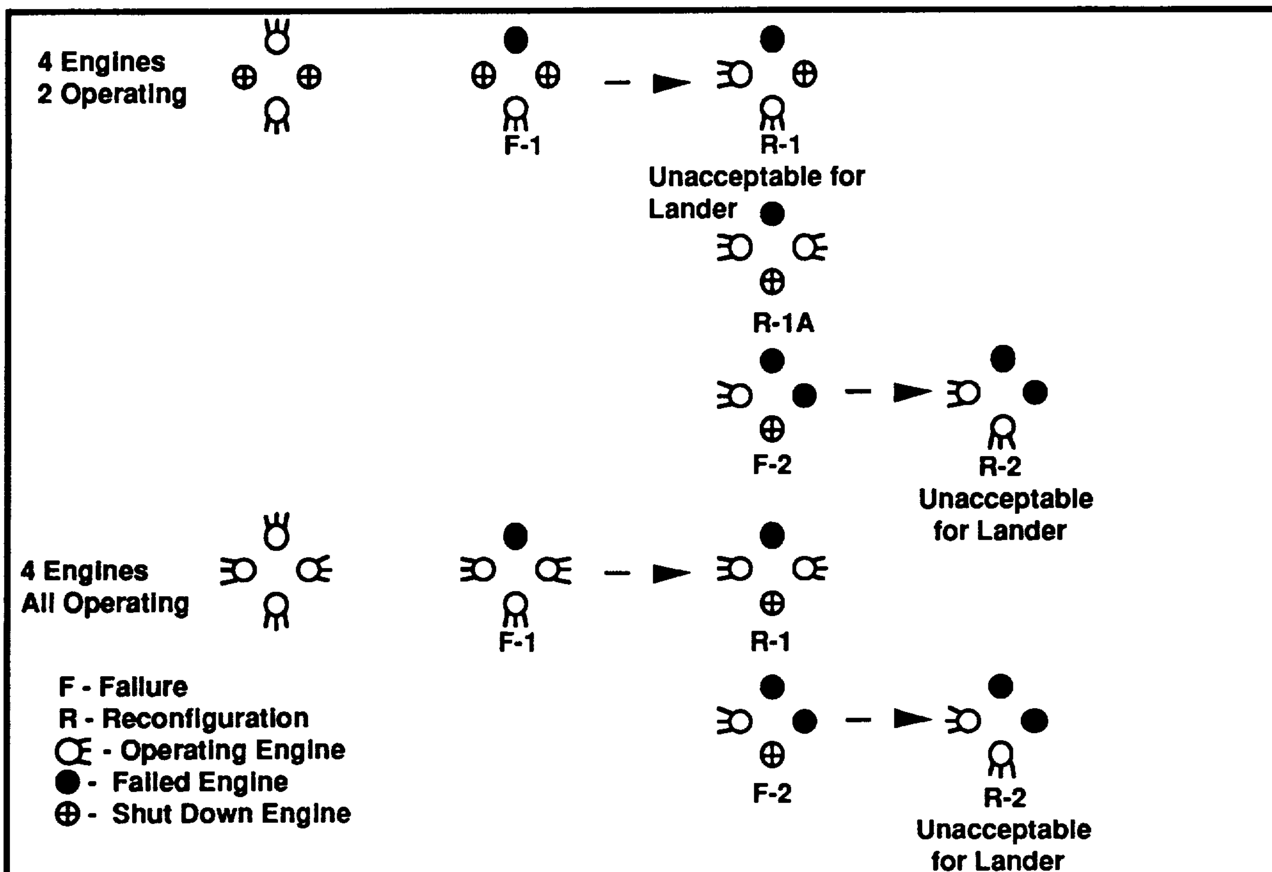


Figure 2.3.4.2.3.3-1 Engine-Out Strategy, 4 Engines, 2 Failures

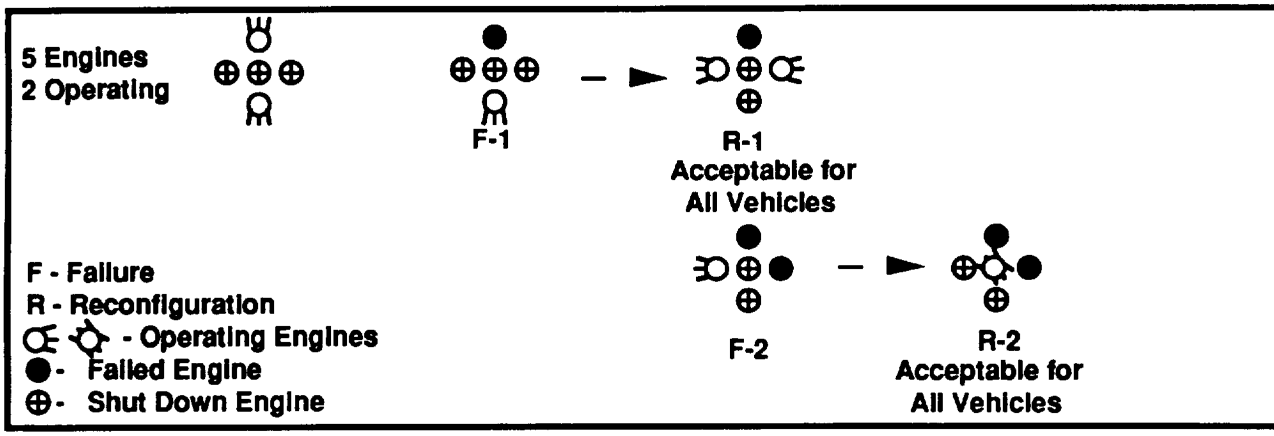


Figure 2.3.4.2.3.3-2 Engine-Out Strategy, 5 Engines, 2 Failures

If the five-engine set is intended to have all engines operating simultaneously, the reconfiguration logic follows a different path, but the results accomplish the same end, as shown in Figure 2.3.4.2.3.3-3. Depending on which engine is arbitrarily selected as the failed engine, either a single engine in the center, or dual symmetrical outboard engines will be left operating after two failures.

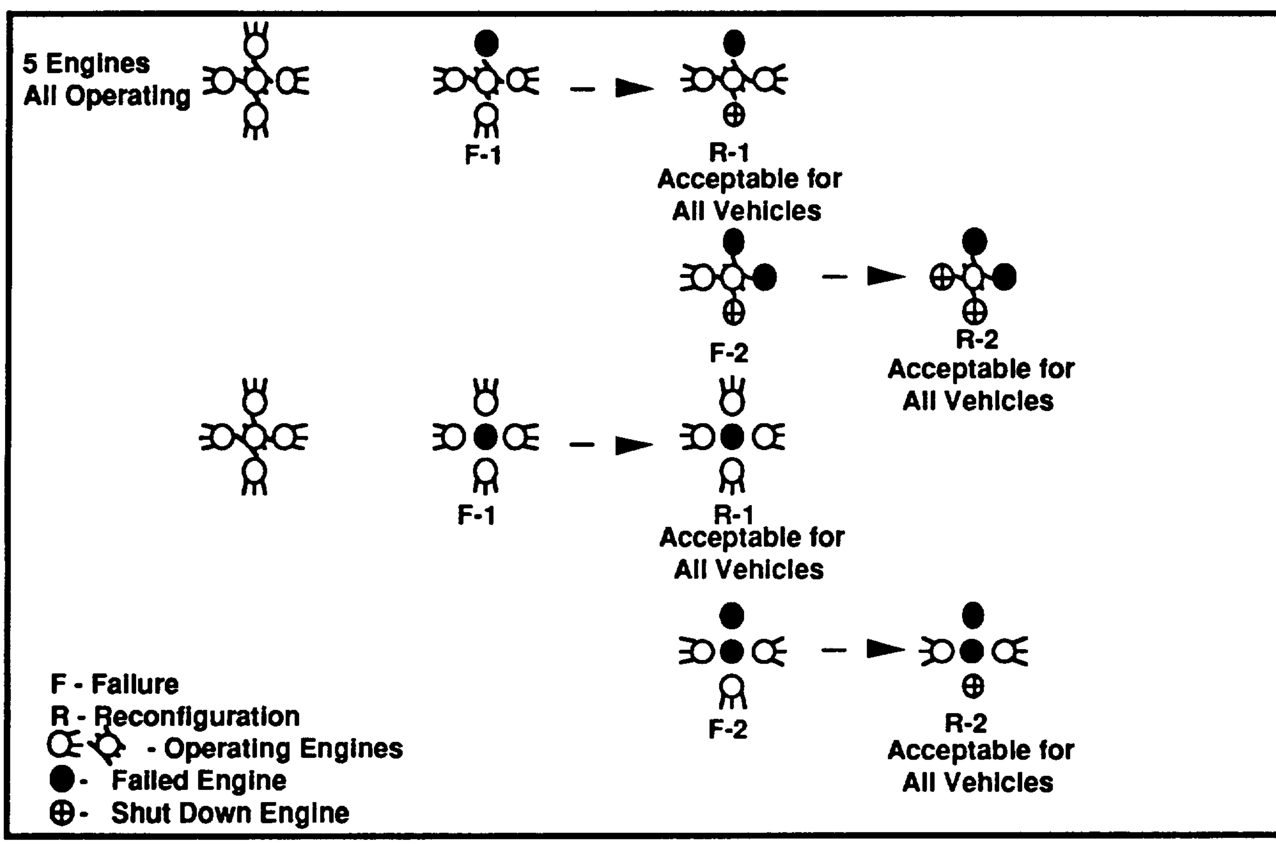


Figure 2.3.4.2.3.3-3 Engine-Out Strategy-5 Engines, 2 Failures (Cont)

2.3.4.2.3.4 Engine System Reliability

One of the aspects of selecting the number of engines to be used is reliability. While this is not the only measure of our ability to count on an engine system to perform its function, it is a system whereby numbers can be assigned which will at the very least influence selections between options. A set of data is presented which allows the comparison of engine set reliability as a function of the presumed engine-out capability of the system.

The Multiple Engine Out Reliability study examines the impact on propulsion system reliability of a multiple-engine configuration, assuming engine out capability as appropriate to provide failure tolerance. The configurations examined consist of two to eight engines. The analysis employs the same engine failure rate throughout. Variation in thrust level is assumed to result in equal total burn times independent of engine configuration. For purposes of the comparison, each single engine propulsion system is assumed to have a reliability of 0.99.

The maintaining of centerline thrust is an assumed requirement, to allow for proper vehicle landing orientation. Ten percent of engine failures are assumed to be catastrophic, based on a presentation by Pratt & Whitney to the Advanced Launch System program. Given a non-catastrophic engine failure, the requirement to maintain centerline thrust is met by shutdown of the symmetrically opposing engine. Probability of successful shutdown is assumed to be 0.999. The two-engine configuration does not allow for one engine out, since failure of one engine results in asymmetrical thrust. The three-engine configuration is linear allowing all engines to operate successfully or an outer engine to fail non-catastrophically, the other outer engine to be detected as non-failed and successfully shutdown, or the center engine to fail non-catastrophically and the outer two engines to operate successfully. Each of the higher-numbered configurations, four to eight engines, is analyzed so that non-catastrophic engine failure, followed by shutdown of the opposing engine as necessary, similarly maintains the centerline thrust. The four-engine configuration boxes the engines in opposing pairs. The five-engine configuration is composed of a four-engine box with a fifth engine in the center. The six-engine and eight engine configurations are ringlike. The seven-engine configuration is composed of a six-engine ring, with the seventh engine in the center.

Results are shown in Figure 2.3.4.2.3.4-1. This figure demonstrates that engine out capability provides a significant increase in the propulsion system reliability for multiple engine configurations.

Engine out capability provides a significant increase in the propulsion system reliability for multiple engine configurations. In Figure 2.3.4.2.3.4-2, the upper right-hand corner of Figure 2.3.4.2.3.4-1 is expanded to enable the associated increases in reliability to be further compared. For the configurations capable of two engines out, allowance of the second engine out condition provides an observable increase in the propulsion system reliability over single engine out. Additional engine out capability, beyond two engines out, provides no significant additional increase in the propulsion system reliability, given the same assumptions as for Figure 2.3.4.2.3.4-1.

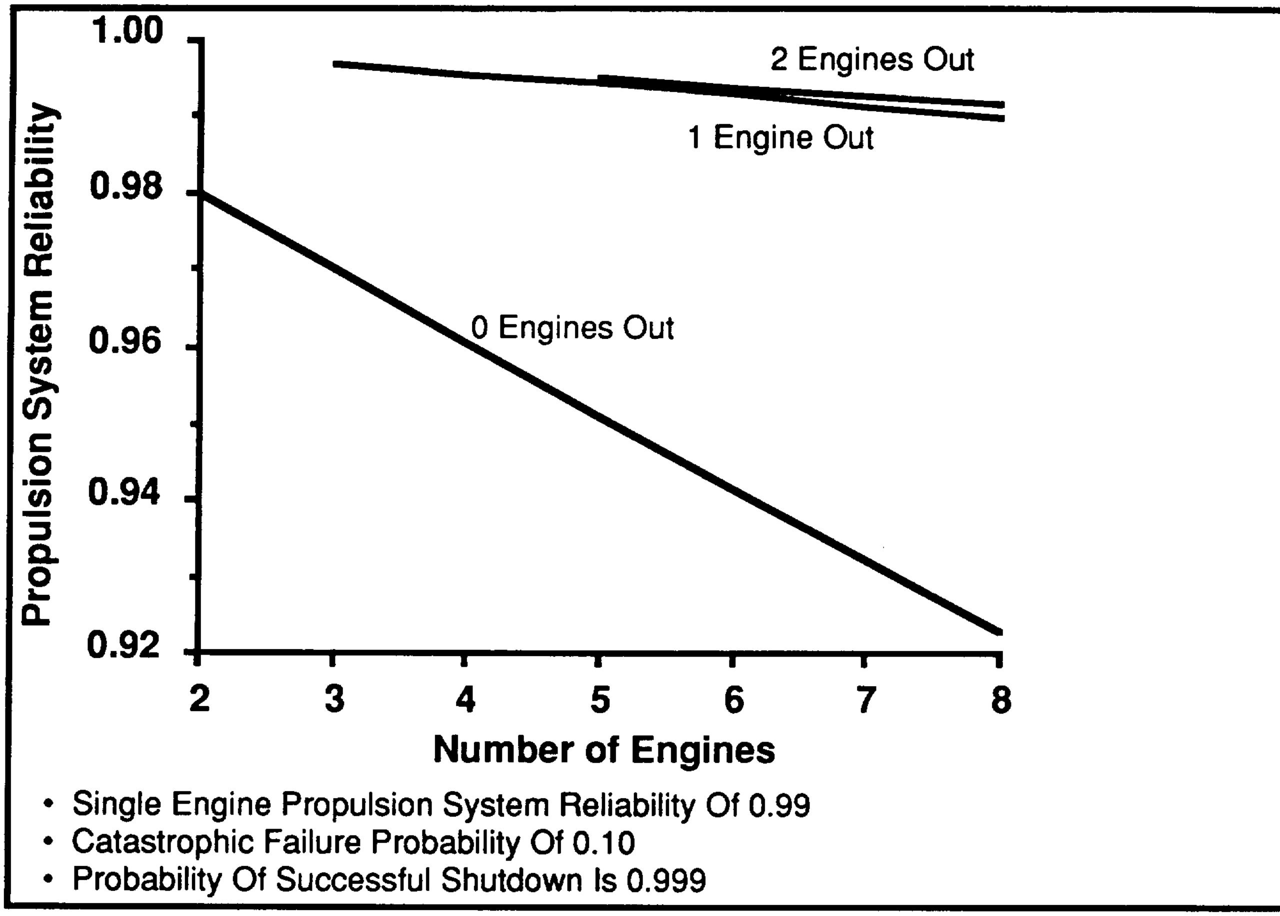


Figure 2.3.4.2.3.4-1 Multiple Engine-Out Reliability

Figure 2.3.4.2.3.4-1 demonstrates the most appreciable gain in overall propulsion system reliability to occur with the increase from zero engines out to a single engine out. Figure 2.3.4.2.3.4-3 depicts an analysis of single engine-out reliability. The assumptions for this chart are the same as those for the multiple engine-out analysis, except that the reliability of each single engine propulsion system is set between 0.99 and 1.00. The reliabilities of the one and two engine propulsion systems, each with zero engines-out, are shown for purpose of reference.

Assuming single engine-out capability, the three-engine configuration is demonstrated to have the greatest propulsion system reliability. Configurations containing additional engines require those engines to run successfully to achieve overall propulsion system success. The reliability of an eight-engine configuration with a single engine-out capability is notably similar to the reliability of a single engine system with zero engines-out. This is a result not only of the eight-engine configuration, but also of the catastrophic failure probability, which is assumed in the analysis to be ten percent.

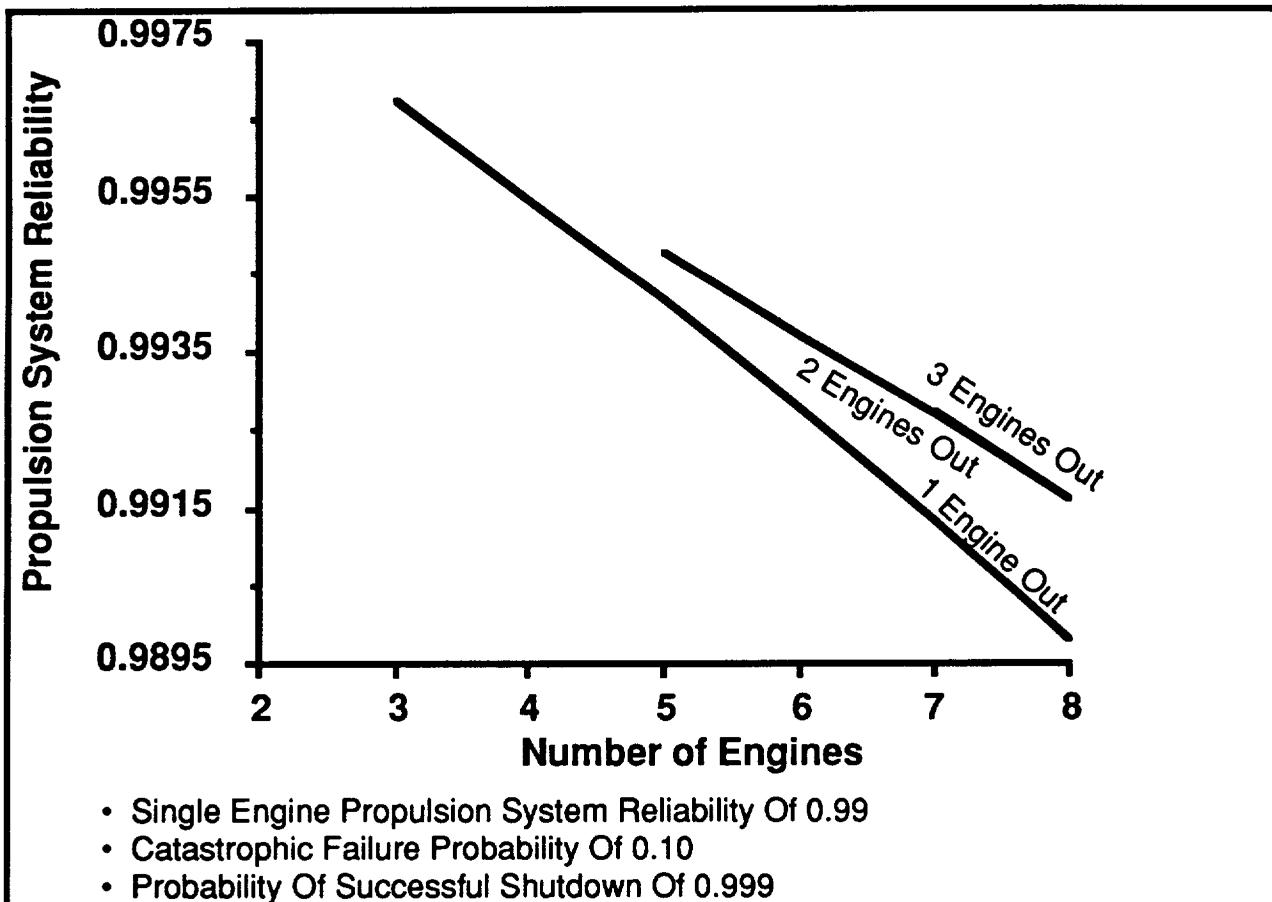


Figure 2.3.4.2.3.4.-2 Multiple Engine-Out Reliability, Expanded

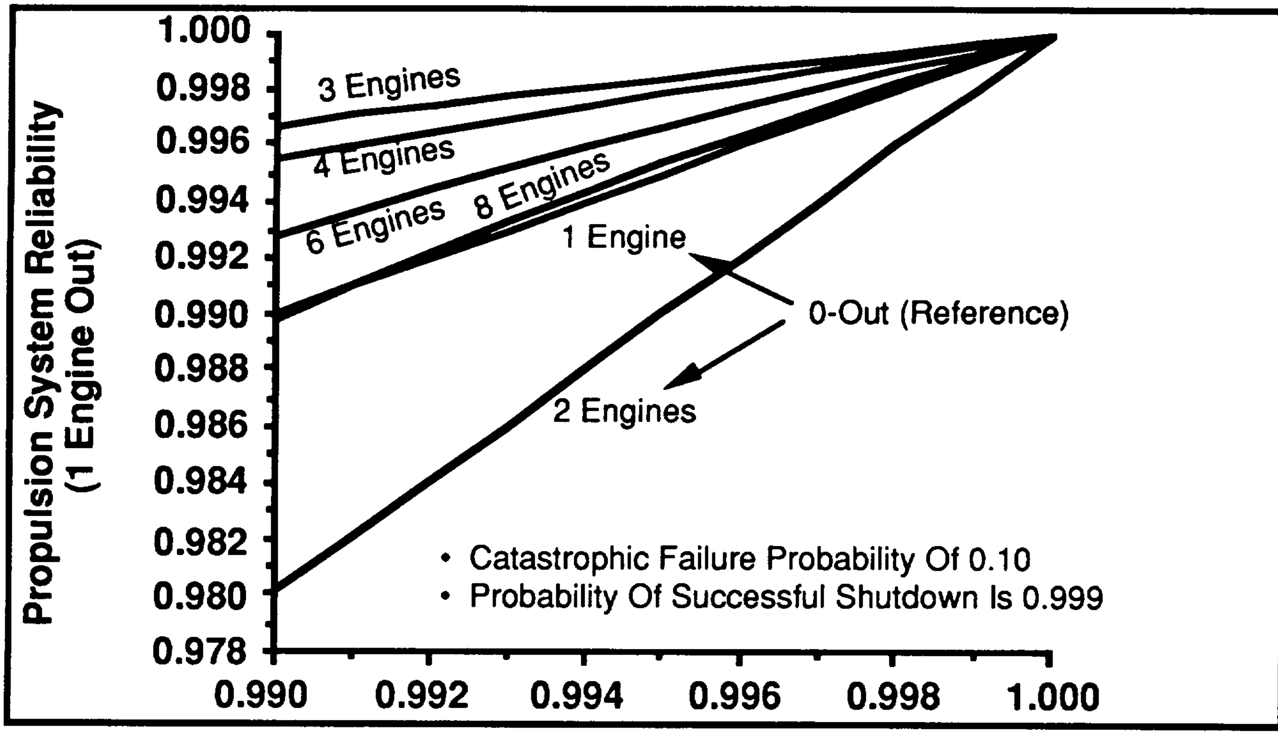


Figure 2.3.4.2.3.4.-3 Single Engine-Out Reliability

For engine configurations using high numbers of engines, single engine-out reliability improves significantly if five percent of engine failures are assumed to be catastrophic rather than ten percent. Thus, it will be worthwhile to identify ways to improve, that is, to decrease, the probability of catastrophic engine failure. One possible means of improvement is the use of shields (i.e. Kevlar) or other devices, to separate the individual engines so that failure of one engine does not result in failure of the overall engine configuration. The assumptions for Figure 2.3.4.2.3.4-4, besides using a catastrophic failure probability of 0.05, are the same as those for the previous single engine-out reliability chart.

2.3.4.2.3.5 LTS Aborts Relative to Number of Engines

During previous evaluations, the number of engines selected appeared to be driven by the requirement for two-fault tolerance, in considering the failure of one entire engine as a fault mode. Tables 2.3.4.2.3.5-1 and 2.3.4.2.3.5-2 attempt to look at the number of engines required from a somewhat different perspective. If one considers the probable course of action due to an engine failure at various stages of a lunar mission, (even though the basic capability to survive and to recover the vehicle has been built into the system), the abort scenarios tend to be somewhat more conservative than if the reliability numbers alone are used to determine when to continue and when to abort a mission. Table 2.3.4.2.3.5-1 was created with the 90 day study vehicle as the reference. The obvious benefit of this vehicle used during a non-steady state mission is the "spare"

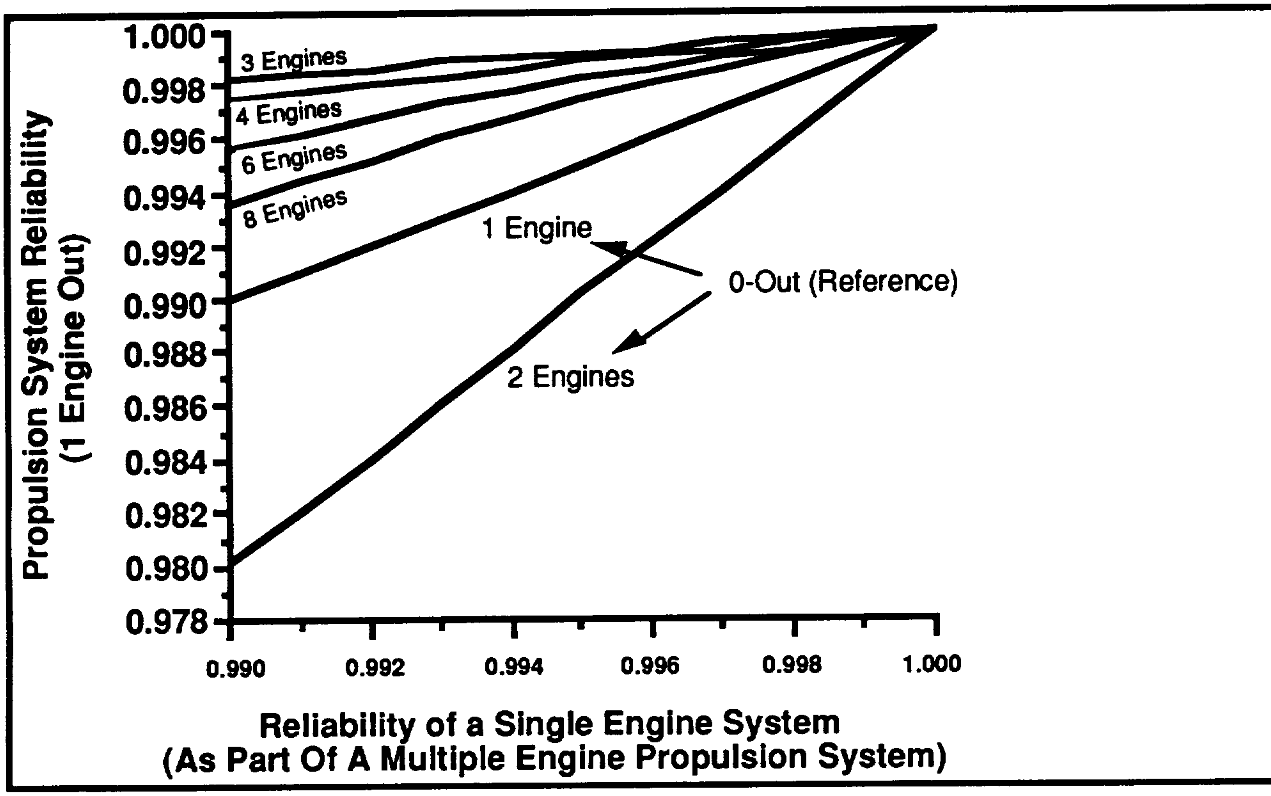


Figure 2.3.4.2.3.4-4 Single Engine-Out Reliability (5% Catastrophic Failure)

Mission Phase	6 Engines		5 Engines		4 Engines		3 Engines		2 Engines	
	Lose 1	Lose 2	Lose 1	Lose 2	Lose 1	Lose 2	Lose 1	Lose 2	Lose 1	Lose 2
Pre TLI Checkout	Abort t Node	Abort to Node	Abort tc Node	Abort t Node	Abort tc Node	Abort t Node	Abort tc Node	Abort t Node	Abort tc Node	Abort to Node (LEV)
During TLI, Early	Compl TLI,Ret to LEO	Compl TLI,Ret to LEO	Compl TLI,Ret to LEO	Compl TLI,Ret to LEO	Compl TLI,Ret to LEO	Compl TLI,Ret to LEO	Compl TLI,Ret to LEO	Compl TLI,Ret to LEO	Compl TLI,Ret to LEO	Abort to Elliptic EO
During TLI, Late	Compl TLI,Ret to LEO (A/B)	Compl TLI,Ret to LEO (A/B)	Compl TLI,Ret to LEO (A/B)	Compl TLI,Ret to LEO (A/B)	Compl TLI,Ret to LEO (A/B)	Compl TLI,Ret to LEO (A/B)	Compl TLI,Ret to LEO (A/B)	Compl TLI,Ret to LEO (A/B)	Compl TLI,Ret to LEO (A/B)	Compl TLI W/RCS Ret to LEO(A/B)
During LOI	Cont	Cont	Cont	Cont	Cont	Compl LOI,Re to LEO	Cont	Compl LOI,Re to LEO	Compl LOI,Ret to LEO	Ellip. LO Ret to EO (LEV)
During Lunar Descent	Land	Abort	Land	Abort	Land	Abort	Abort	Abort	Abort	Fail
During Lunar Ascent	Cont	Cont	Cont	Cont	Cont	Cont	Cont	Cont	Cont	Fail
During TEI	Cont	Cont	Cont	Cont	Cont	Cont	Cont	Cont	Cont	RCS

Assumes Multiple Propulsion Systems (90 Day Study Vehicle)

Options Reflect Initial Flight Only. LTV and LEV are mated During Transfer from Earth to Moon

The Number of Engines Relates to The Vehicle (LTV/LEV) in use During the Specific Mission Phase

Figure 2.3.4.2.3.5-1 LTS Aborts Relative to Number of Engines (90 Day Vehicle) propulsion system on the LEV. This advantage, however, goes away when the LTV is used alone to transfer cargo or men to lunar orbit. Many of the abort decisions shown here are arbitrary. Several of them could be influenced by the national political mood at the time relating to the risks to be taken with astronauts. In general, however, a conservative approach is shown, which is likely to be the case.

Figure 2.3.4.2.3.5-2 presents the abort decisions relative to numbers of engines and loss of one or two of these relative to the preferred vehicle concept, 4E-5B. There are a number of situations that would likely result in a different decision, as shown by the boxed areas on the figure.

Mission Phase	6 Engines		5 Engines		4 Engines		3 Engines		2 Engines	
	Lose 1	Lose 2	Lose 1	Lose 2	Lose 1	Lose 2	Lose 1	Lose 2	Lose 1	Lose 2
Pre TLI Checkout	Abort t Node	Abort to Node	Abort tc Node	Abort tc Node	Abort tc Node	Abort t Node	Abort to Node	Abort t Node	Abort t Node	RCS Abort to Node
During TLI, Early	Abort t Elliptic EO	Abort to Elliptic EO	Abort tc Elliptic EO	Abort tc Elliptic EO	Abort tc Elliptic EO	Abort t Elliptic EO	Abort to Elliptic EO	Abort t Elliptic EO	Abort tc Elliptic EO	Abort to Elliptic EO
During TLI, Late	Compl TLI,Ret to LEO (A/B)	Compl TLI,Ret to LEO (A/B)	Compl TLI,Ret to LEO (A/B)	Compl TLI,Ret to LEO (A/B)	Compl TLI,Ret to LEO (A/B)	Compl TLI,Ret to LEO (A/B)	Compl TLI,Ret to LEO (A/B)	Compl TLI,Ret to LEO (A/B)	Compl TLI,Ret to LEO (A/B)	Compl TLI W/RCS Ret to LEO(A/B)
During LOI	Compl LOI,Re to LEO	Compl LOI,Ret to LEO	Compl LOI,Ret to LEO	Compl LOI,Ret to LEO	Compl LOI,Re to LEO	Comp LOI,R to LEC	Compl LOI,Ret to LEO	Compl LOI,Re to LEO	Compl LOI,Re to LEO	Probabl Failure
During Lunar Descent	Land	Abort	Land	Abort	Land	Abort	Abort	Abort	Abort	Failure
During Lunar Ascent	Cont	Cont	Cont	Cont	Cont	Cont	Cont	Cont	Cont	Failure
During TEI	Cont	Cont	Cont	Cont	Cont	Cont	Cont	Cont	Cont	RCS

Single Propulsion
(4E-5B Vehicle)

☐ Represents Reduced Capability Compared with 90 Day Study Vehicle

Figure 2.3.4.2.3.5-2 LTS Aborts Relative to Number of Engines (4E-5B Vehicle)

There are a number of conclusions that can be drawn from this evaluation, (which was created by addressing mission continuation options in the event of one or two engines out). Most of the conclusions have already been drawn from other evaluations, but some are worth repeating. There do not appear to be any discriminators from the abort consideration between five or six engines. There are however, discriminators in the abort sense when less than five engines are used. The selection between five and six engines must be made in consideration of other items, for instance, the desire to have centerline thrust during lunar landing and takeoff in the event two engines have failed. This will simplify and perhaps even negate the need for engine gimbals. Placement of the engines, the philosophy for their use, line routing and other configuration influences all play a part in the selection. We have selected the five-engine approach previously due in a large part to the desire for centerline thrust with two engines out. This evaluation helps to reinforce that decision. A summary of the conclusions is presented in Table 2.3.4.2.3.5-1.

Table 2.3.4.2.3.5-1 Summary of Conclusions for Configuration Evaluation Relative to Aborts

- **When Vehicle Configuration Is Considered with Regard for Its Effect on Mission Abort Scenarios, Different Conclusions Surface Compared to Configurations Based on Redundancy Requirements Alone**
- **There Is a Significant Advantage in the Ability of a Dual Propulsion System Configuration (90 Day Vehicle) to Complete a Non-Steady State Lunar Mission over a Single Propulsion System Vehicle (4E-5B), Regardless of the Number of Engines Selected. This Assumes that Both Systems Have Two-Failure Tolerance.**
- **There Are No Abort Discriminators between Five Engines and Six Engines**
- **There Are Abort Discriminators Using Three or Four Engines in Either the 4E-5B or 90 Day Configuration**
- **Use of Two Engines Does Not Provide Two-Failure Tolerance, and Indicates Loss of Mission for a Number of Scenarios**
- **Given the Above, and In Consideration of the Balanced Centerline Thrust Vector in the Event of Two Engine Failures During Lunar Landing or Takeoff (Which Also may Make Thrust Vector Control Feasible and Allow Removal of the Gimbal System), Five Engines Still Appears to be the Appropriate Number**

2.3.4.2.4 Engine Selection Parametrics

A number of general engine parametrics are presented in section 2.3.4.2.1. The following sections contain parametrics that allow engine selection for specific vehicles.

2.3.4.2.4.1 Single Engine Thrust for LTV/LEV

This section evaluates the requirements for engine selection for TLI and lunar landing phases of a lunar mission, with the goal of selecting a common engine to best accomplish both objectives. The

TLI phase is important because of the large gravity losses encountered if the thrust to weight ratio is too low. The lunar landing phase is important because of the need to throttle the engines to allow hover and descent in the lunar gravity. This analysis was performed using a "90 day study configuration" as the baseline. It leads to slightly different results than an analysis presented later in this section that deals with the 4E-5B configuration.

Analysis was performed to determine the gravity losses incurred by the lunar transfer vehicle (LTV) on the trans-lunar injection (TLI) burn for various engine thrust levels. The results of this analysis are shown in Figure 2.3.4.2.4.1-1 for differing numbers of LTV engines. Note that the IMLEO's remained constant and that increased gravity losses generally result in decreased cargo masses. However, since more or larger engines have more mass, the largest cargo masses are not necessarily achieved by just going to more or larger engines. Their lower gravity losses and added mass must be traded off against each other to determine the number of engines and their thrust level to maximize the lunar cargo.

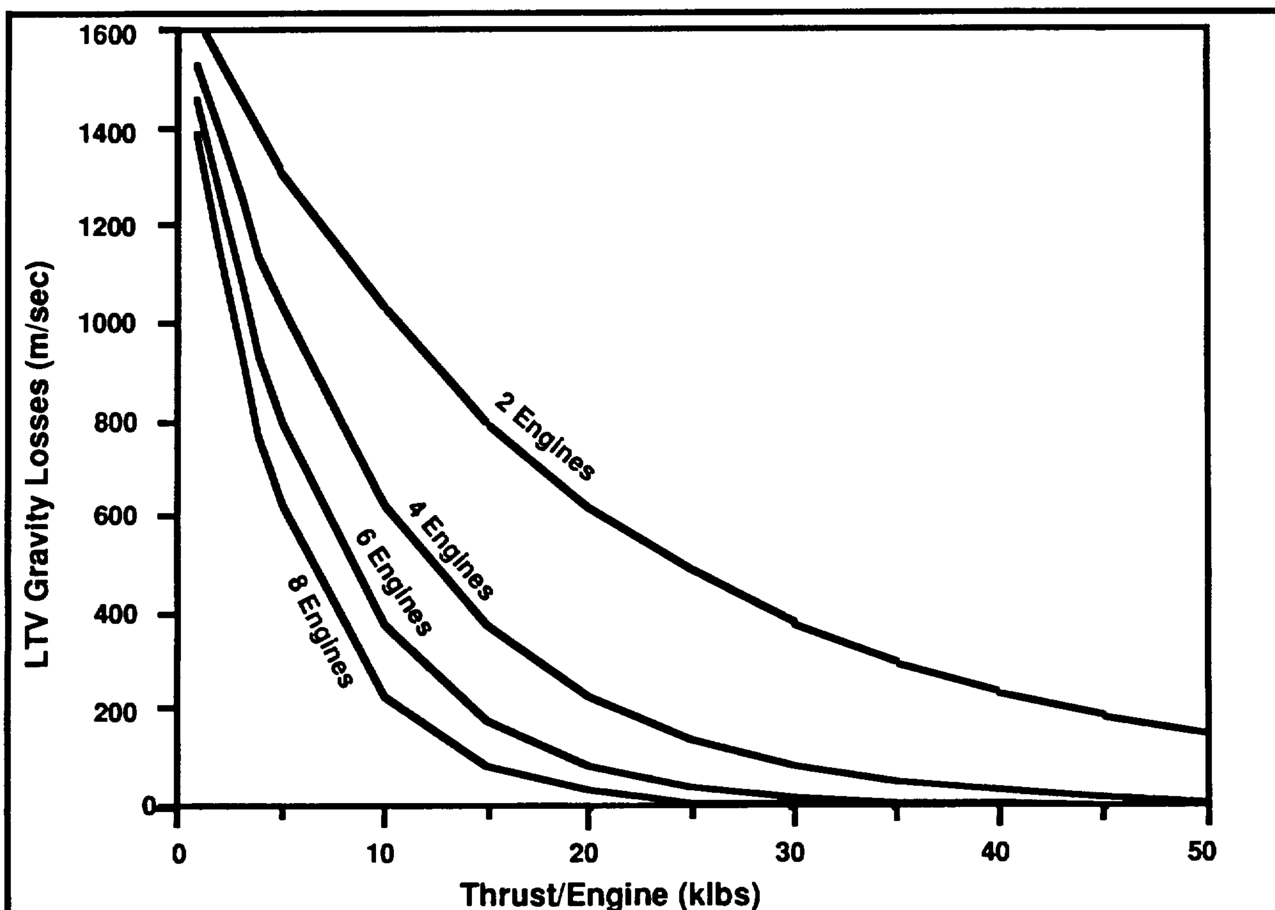


Figure 2.3.4.2.4.1-1 TLI Gravity Losses vs Thrust per Engine for 1 Burn

Figure 2.3.4.2.4.1-2 presents the gravity losses that are incurred when a two burn strategy is used by the LTV to achieve the proper trans-lunar trajectory. When compared to the losses that are associated with the one burn strategy, it can be seen that the two burn method is substantially more efficient in terms of Δ-velocity. However, the two burn method has some other drawbacks that cannot be deduced from the figure. For example, the two burn case has a longer mission duration, will require more tank insulation (or increased boiloff), and will have a different radiation environment.

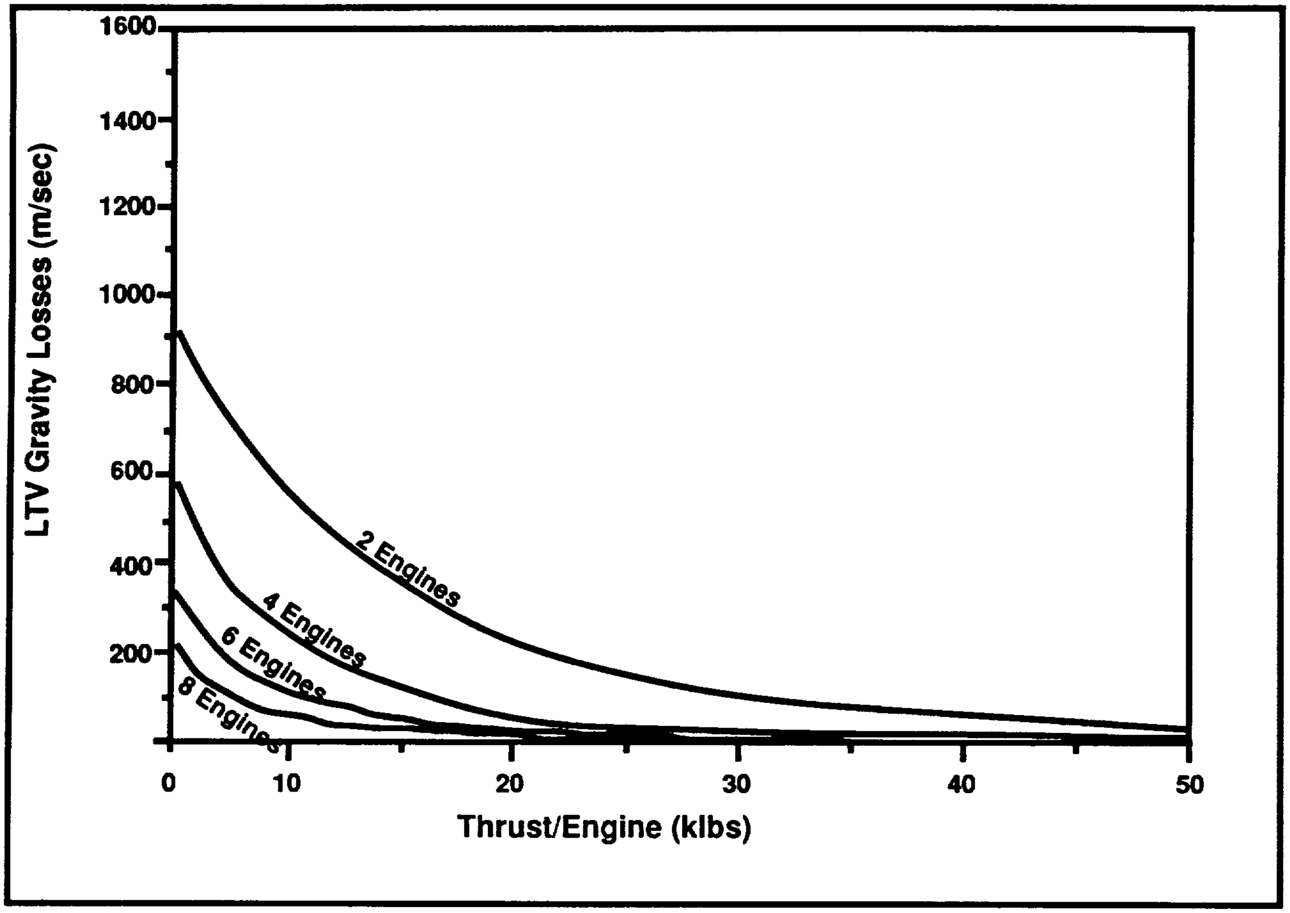


Figure 2.3.4.2.4.1-2 TLI Gravity Losses vs Thrust per Engine for 2 Burns

Another parameter that must be considered if a common engine is to be used on the LTV and lunar Excursion Vehicle (LEV) is the throttling ratio of the engine. Figure 2.3.4.2.4.1-3 shows the results of analysis that was done to determine the throttling ratio required, depending on both the thrust of the engine and the number selected for the LEV. This analysis assumes a 100% hover capability, although more recent analyses have shown that 80% lunar hover thrust may be a more appropriate parameter.

If the previous two figures are combined, some evaluation may be made of the optimum number of engines and thrust and throttling levels that may be desired to have a single engine capable of satisfying both the lunar transfer and lunar landing operations. This is presented in Figures 2.3.4.2.4.1-4 and -5. The range of 2 to 8 LTV engines has been shaded as the probable selection range. Similarly, the range of 2 to 6 LEV engines and the throttling range of 20:1 and below has been shaded as being the likely range. The throttling ratio limit of 20:1 is believed to be the practical upper limit for a hydrogen/oxygen engine of this type. A throttling ratio of approximately 10:1 is even more desirable as a much lower cost option.

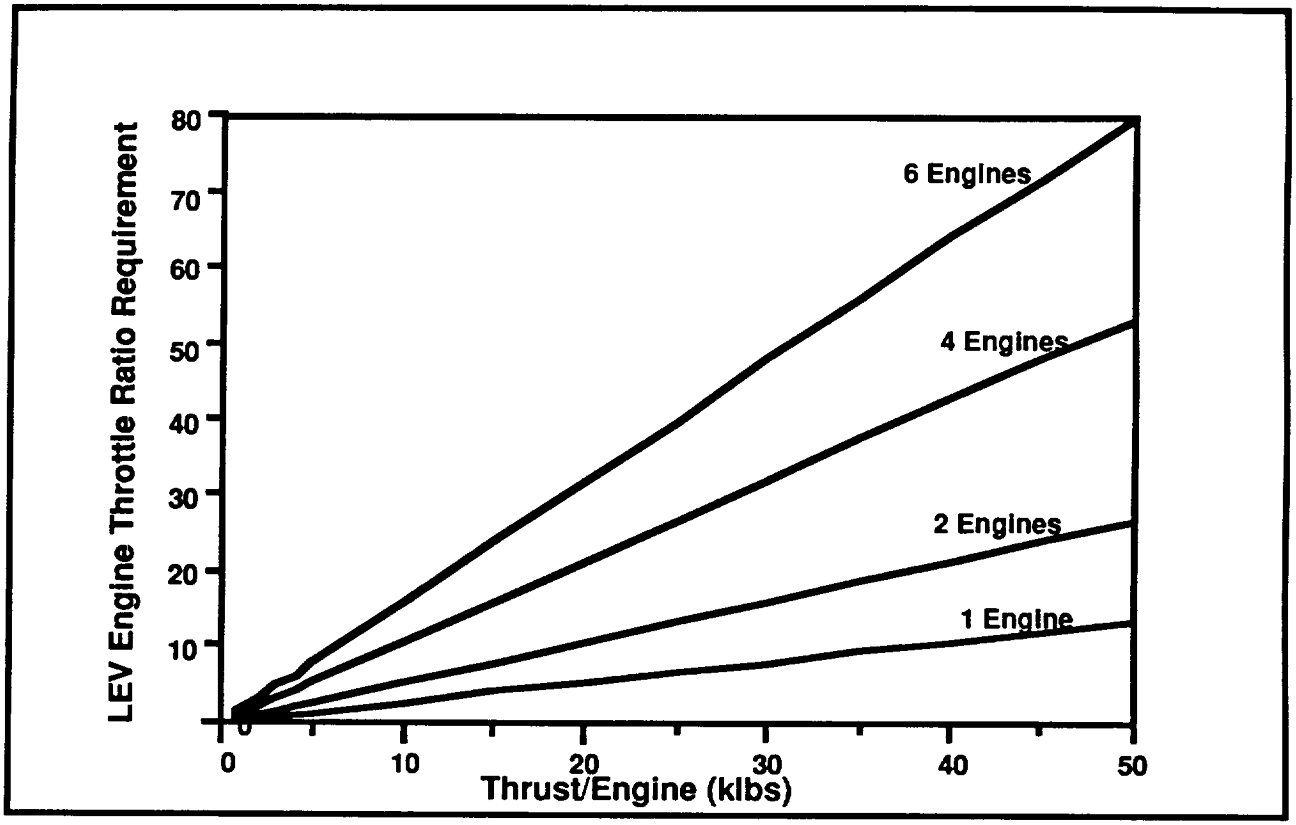


Figure 2.3.4.2.4.1-3 LEV Throttling Requirement vs Thrust per Engine

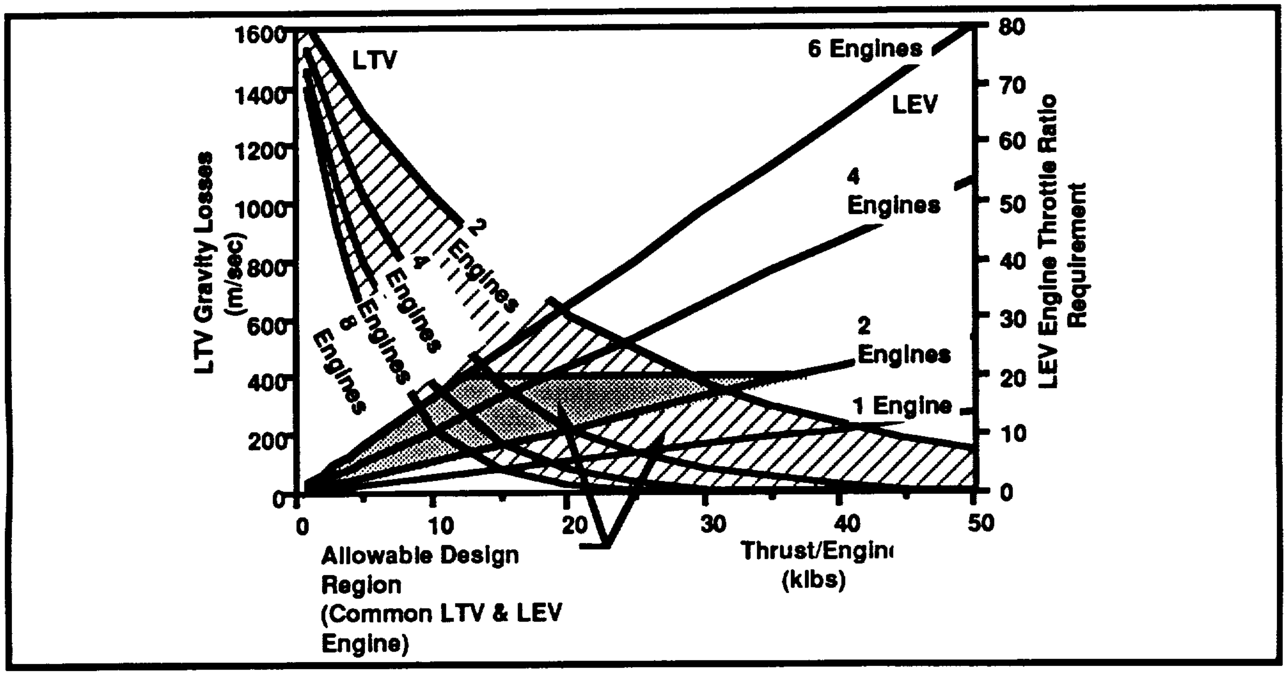


Figure 2.3.4.2.4.1-4 Gravity Losses (1 Burn) and Throttling Requirements

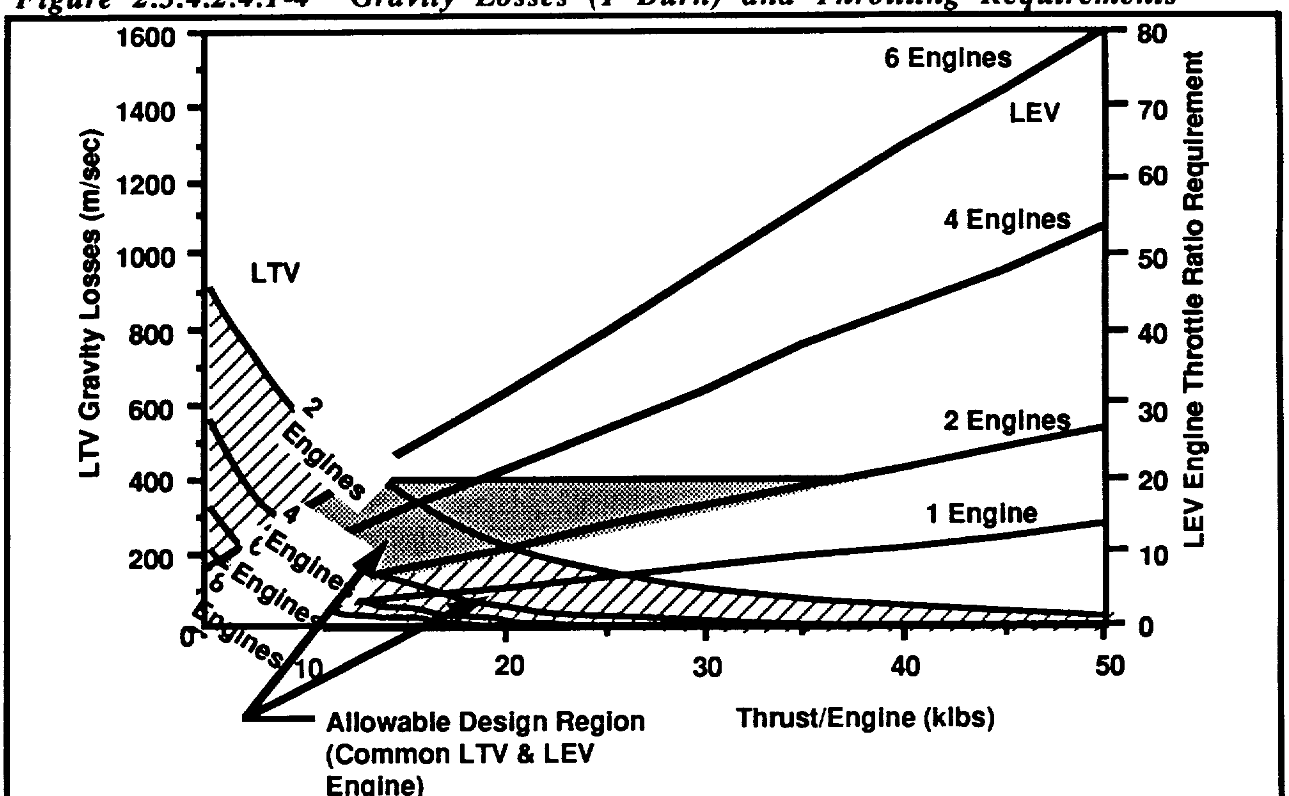


Figure 2.3.4.2.4.1-5 Gravity Losses (2 Burn) and Throttling Requirements

Optimum Thrust and Throttling Requirements vs. Number of Engines - The following four figures encompass the engine performance characteristics relative to the 4E-5B selected vehicle with 158.6 metric tons of propellant. Subsequent refinement of this vehicle increased the quantity of propellant to 174 metric tons and the effect of that increase on engine thrust level will be treated in section 2.3.4.2.4.3. Figure 2.3.4.2.4.2-1 is a plot of the total weight delivered to the lunar surface as a function of engine thrust if only two engines are fitted. The delivered weight reaches a maximum a thrust of 40,000 pounds per engine, although the curve is very flat and a thrust of 30,000 pounds per engine delivers nearly the same amount.

The throttling ratio required is also plotted as a function of engine thrust, both for the nominal mission case where the vehicle delivers as much payload as it can, and for a minimum weight mission where no cargo is delivered, just the crew and enough propellant for them to return to earth. The throttling ratio values include the capability to lower the thrust to 80% of the landing weight.

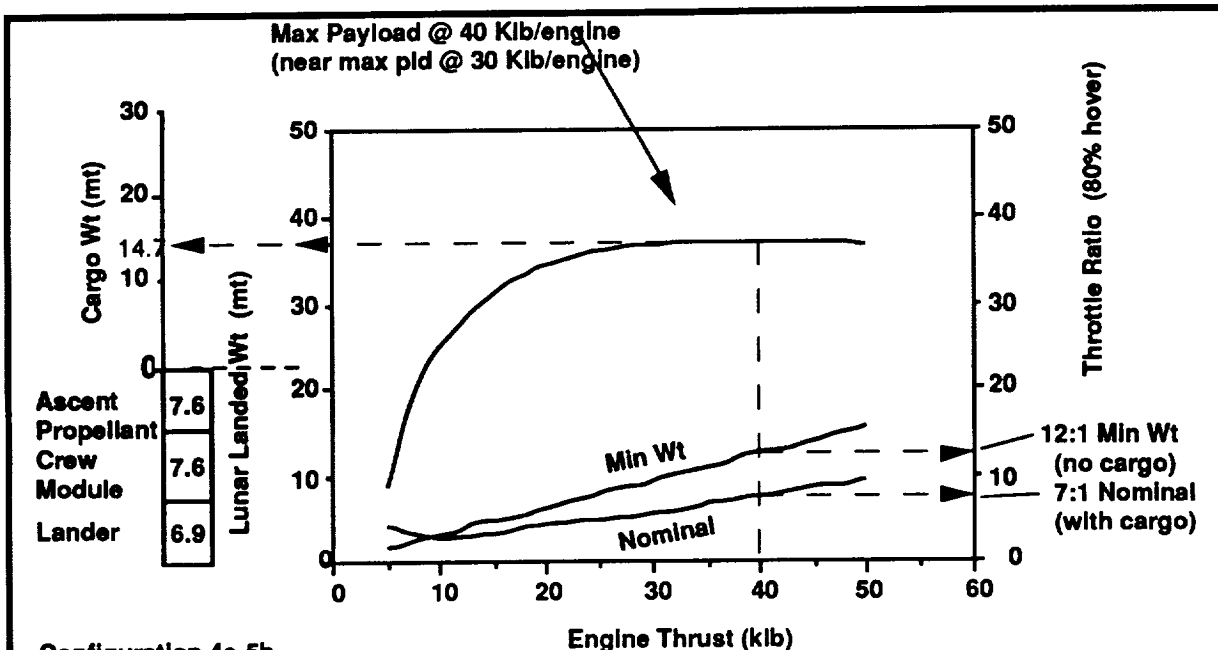
At the indicated thrust value of 40,000 pounds per engine, the required throttle ratio for a nominal mission is 7 to 1, which is very attainable for either an RL-10 derivative or an ASE (although 40,000 pounds is beyond the thrust presently envisioned for these engines). The minimum weight mission needs a somewhat deeper throttling ratio of about 12 to 1, but this is not an unreasonable requirement.

Figure 2.3.4.2.4.2-1, in combination with the next three charts, will show that the optimum total thrust is approximately the same regardless of the number of engines, if the propulsion system thrust-to-weight ratio is about the same for all configurations. Since the total thrust is the same, the throttling ratio requirements are also the same.

Figure 2.3.4.2.4.2-2 shows that for four engines, the optimum engine thrust level is 20,000 pounds.

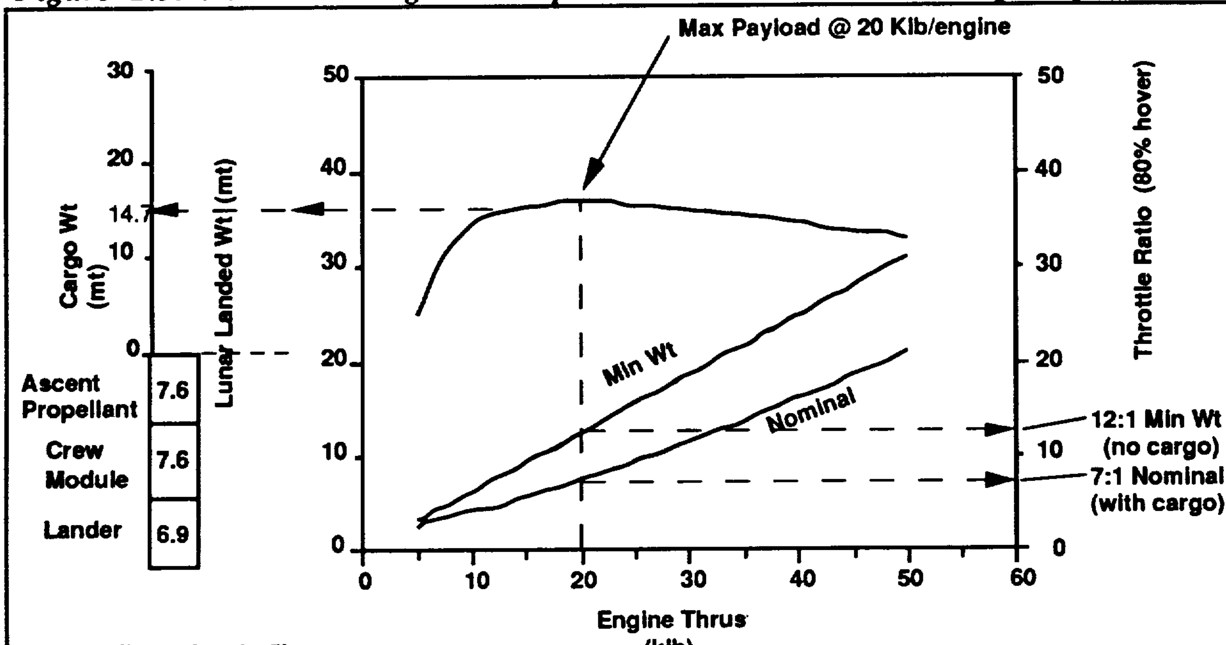
Figure 2.3.4.2.4.2-3 shows that for six engines, the optimum engine thrust level is 12,000 pounds.

Figure 2.3.4.2.4.2-4 shows that for eight engines, the optimum engine thrust level is 10,000 pounds.



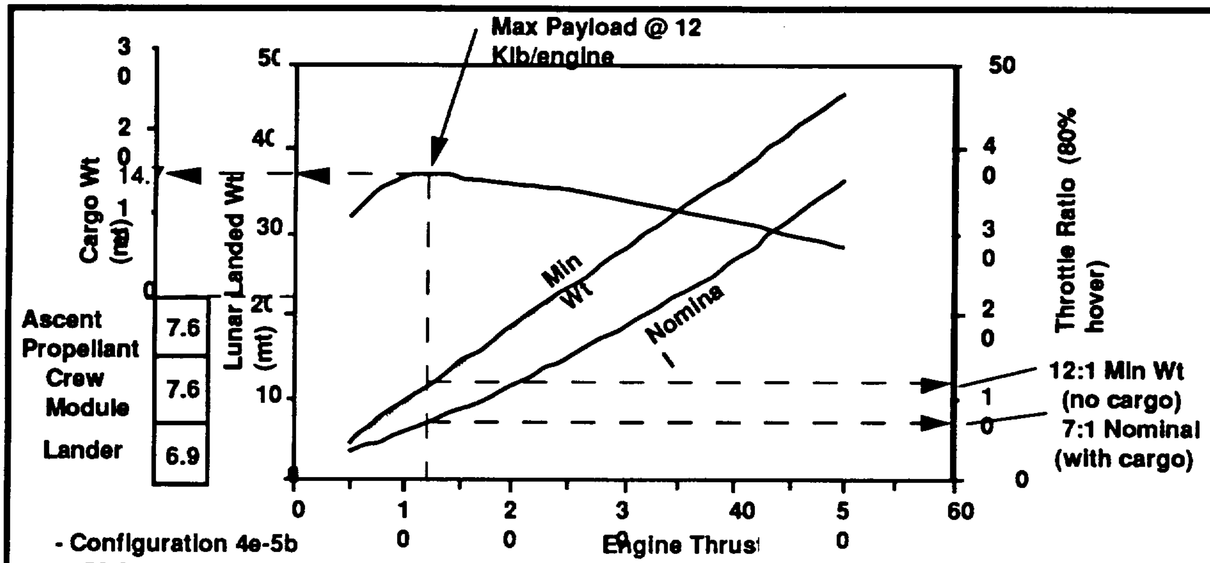
- Configuration 4e-5b
- 158.6 mt total propellant
- propulsion system thrust/weight = 23
  (3460 lb prop. sys. wt. - includes engines, gimball system, feedlines, valves, PU system, press. system, RCS system.)
- Lunar landed weight optimum at approx. 70-80 klb thrust regardless of # of engines

Figure 2.3.4.2.4.2-1 2 Engines - Optimum Thrust and Throttling Requirements



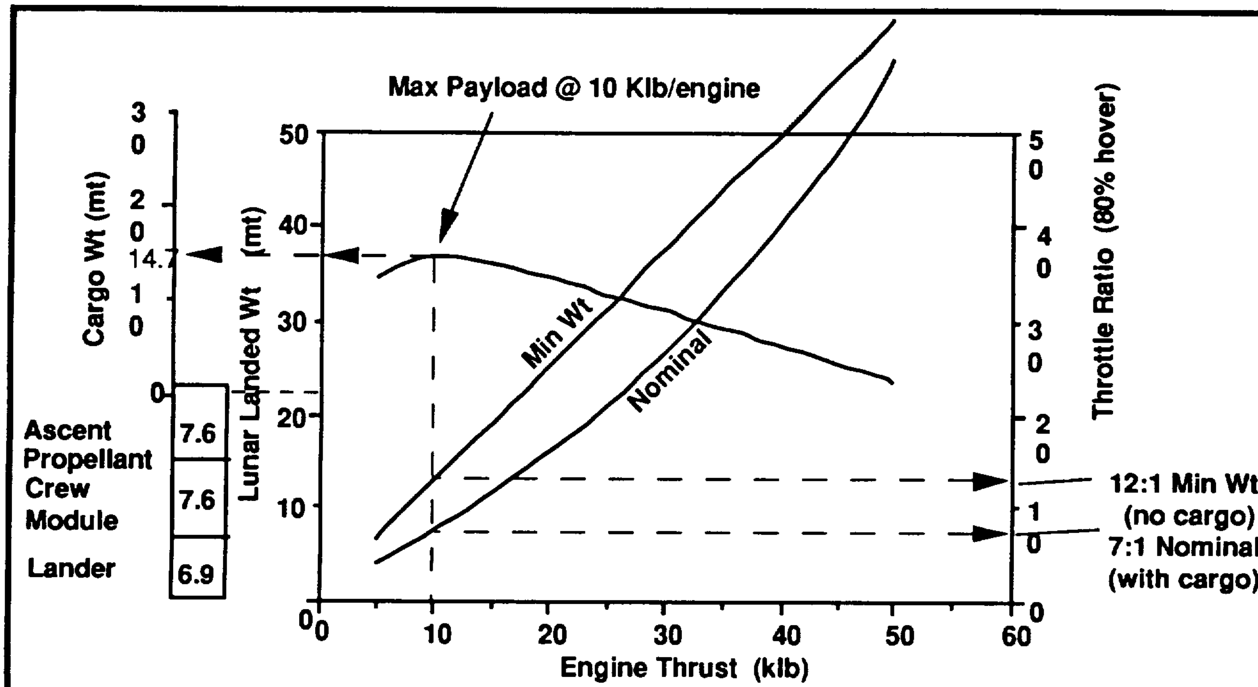
- Configuration 4e-5b
- 158.6 mt total propellant
- propulsion system thrust/weight = 23
  (3460 lb prop. sys. wt. - includes engines, gimball system, feedlines, valves, PU system, press. system, RCS system.)
- Lunar landed weight optimum at approx. 70-80 klb thrust regardless of # of engines

Figure 2.3.4.2.4.2-3 4 Engines - Optimum Thrust and Throttling Requirements



- Configuration 4e-5b
- 158.6 mt total propellant
- propulsion system thrust/weight = 23
  (3460 lb prop. sys. wt. - includes engines, gimball system, feedlines, valves, PU system, press. system, RCS system.)
- Lunar landed weight optimum at approx. 70-80 klb thrust regardless of # of engines

Figure 2.3.4.2.4.2-3 6 Engines - Optimum Thrust and Throttling Requirements



- Configuration 4e-5b
- 158.6 mt total propellant
- propulsion system thrust/weight = 23
  (3460 lb prop. sys. wt. - includes engines, gimball system, feedlines, valves, PU system, press. system, RCS system.)
- Lunar landed weight optimum at approx. 70-80 klb thrust regardless of # of engines

Figure 2.3.4.2.4.2-4 8 Engines - Optimum Thrust and Throttling Requirements

4E-5B Engine Thrust Optimization - Final refinement of the performance parameters of the 4E-5B vehicle increased the quantity of propellant to 174 metric tons. The effect of that increase on engine thrust optimization is treated here. Figure 2.3.4.2.4.3-1 is similar to those previously presented to assist in selection of number of engines, engine thrust, and throttling ratio. It has been updated to reflect the current weights and performance predictions for the 4E-5B vehicle concept. Only the 6 Engine chart is shown here as an example, even though charts were previously presented for 2, 4, 6, and 8 engines. Although the weight of the vehicle has increased somewhat, the throttling ratio required for lunar landing and ascent has not changed from the 7:1 and 12:1 determined before. The most efficient engine thrust for this case has increased from 12,000 lb to 14,000 lb, due to the vehicle weight increase during its optimization.

Figure 2.3.4.2.4.3-2 represents the optimum thrust level per engine, depending on the number of engines selected for the 4E-5B vehicle. It assumes 14.6 tonnes of lunar cargo is to be delivered on a piloted mission. It also shows the variation in thrust level that is possible, with a corresponding

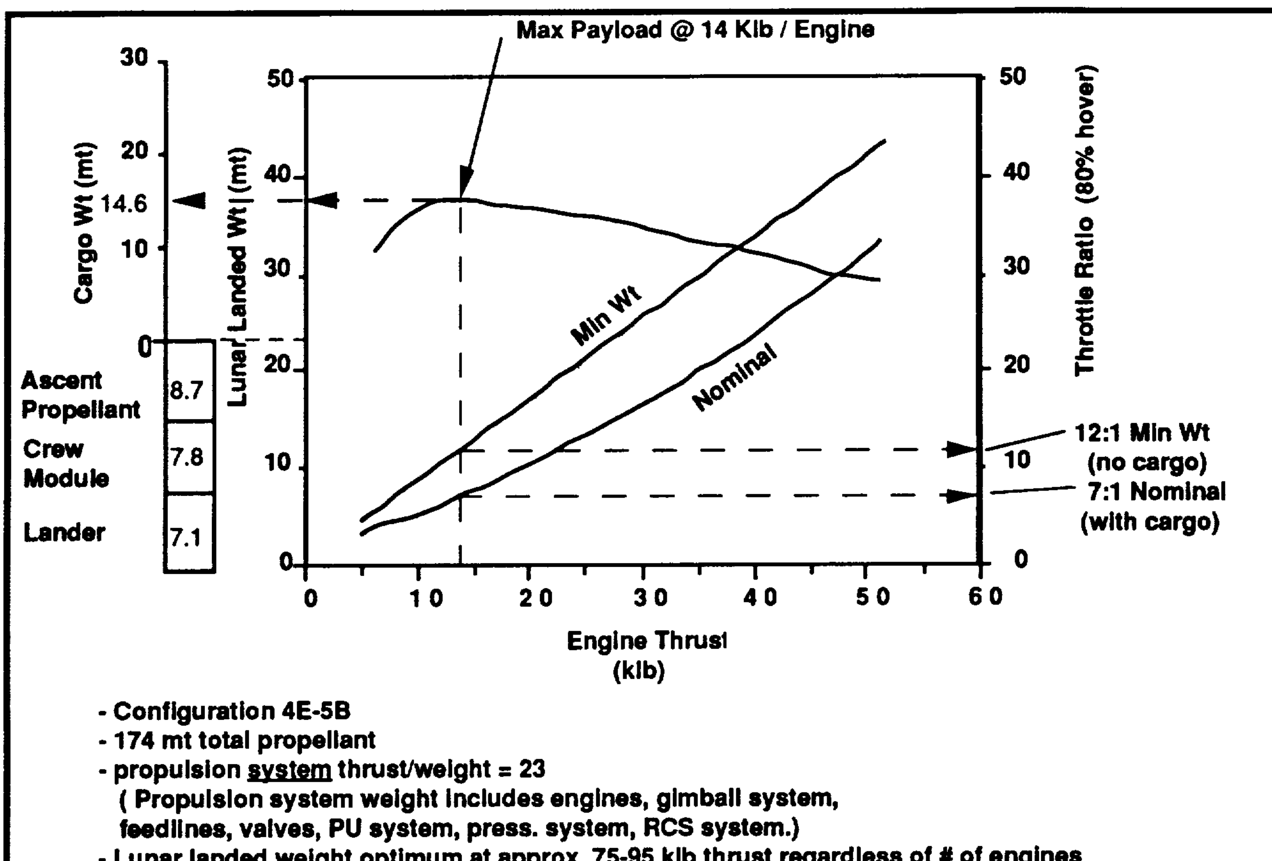


Figure 2.3.4.2.4.3-1 Typical Trust and Throttling Parametrics (6 Engines) for 174 mt Total Propellant

loss (arbitrarily 10%) in cargo carrying capability. For the present vehicle, 5 engines have been selected. The optimum thrust for a 5 engine configuration is nearly 18,000 lb, but a thrust level from 12,000 to 28,000 lb would still be fairly reasonable. As will be seen in the configuration description portion of this report, the actual engine thrust that has been selected for the 4E-5B configuration is approximately 20,000 lb. This would seem to be somewhat higher than optimum, but there are several factors influencing this slight increase: 1), there are more data generally available for engines in this range; 2), it allows for potential weight growth; 3), it allows for a return to a 4 engine configuration, should the MACE requirements and landing logic be reassessed in favor of 4 engines; and 4), it defines the potentially largest engine layout area, so that any subsequent changes in engine architecture should have no effect on the vehicle size.

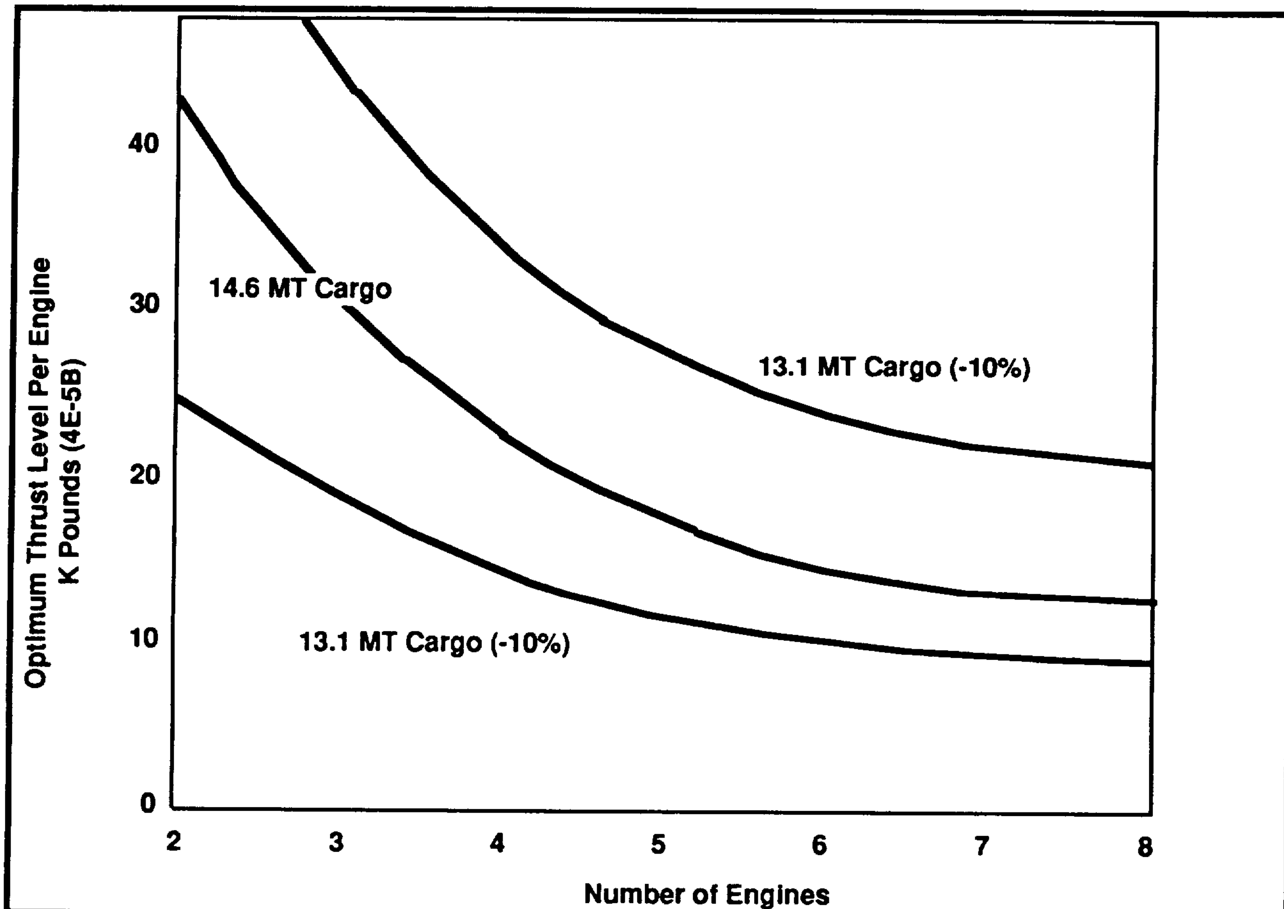


Figure 2.3.4.2.4.3-2 Optimum Thrust per Engine VS Number vs Engines (4E-5B)

Lunar Throttling Range Required - To select the thrust level of the engine system to be used, it is important to determine the necessary thrust as a percentage of the thrust necessary to hover in the lunar gravity environment. If the thrust level never goes below 100% of that needed to hover, then it is not possible to hover and to subsequently begin another descent. It is easy to descend from a hover if the thrust may be reduced to a low value, i.e. 50% of lunar thrust, but this may require an engine throttling ratio more severe (and expensive) than necessary. For this evaluation, thrust and weight data were obtained from the Apollo landings, and evaluated to determine the lowest thrust level obtained. Viking and Surveyor missions were also evaluated, but the actual data were not available from those missions, only the design requirements. Evaluation of the Apollo missions reveals that the thrust varied from 75% to 112% of lunar hover. This is a fairly wide dispersion, and is probably significantly wider than would be experienced on a new vehicle, which would incorporate state of the art avionics. Of interest is that Apollo 14 and 15 apparently landed while continuing to decelerate. In reviewing these data, an engineering judgment was made that an appropriate preliminary design point is 80 % of lunar hover thrust. These data are shown in Figure 2.3.4.2.4.4-1. This number should be used for engine thrust and throttling evaluations, although it is subject to revision during subsequent programs when flight operations scenarios and avionics systems become better defined. It is anticipated that the 80% number is somewhat conservative, and that subsequent revisions will likely be upward.

Starting with the requirement for 80% of lunar hover thrust for lunar landing, the throttling ratio required of the engines is shown in Figure 2.3.4.2.4.4-2 relative to the number of engines operating. This assumes the 4E-5B configuration, and an engine thrust level of 20,000 lb each. It also includes engine-out possibilities, by showing throttling ratio per number of engines operating.

- **Throttling Ratio Of Engines Is Determined By Lunar Landing Requirement**
- **Thrust Approximately Equal To Lunar Weight Or Less Is Required To Land**
- **Avionics Fidelity And Engine Response To Throttle Commands Are Factors**
- **Experience (Lowest Total Engine Thrust As % of Lunar Weight)**

Mission	Lowest Tot. Engine Thrust	Lunar Wt. of Veh	%
Viking	Unknown	Unknown	80 *
Surveyor	Unknown	Unknown	80 *
Apollo 11	2520 lb	2650 lb	95 **
Apollo 12	1995 lb	2650 lb	75 **
Apollo 14	2940 lb	2620 lb	112 **
Apollo 15	2950 lb	2913 lb	101 **
Apollo 16	2500 lb	2913 lb	86 **

* Design
** Experienced

- **Recommended Thrust Ratio, Based On Flight Data and Engineering Judgement: 80%**

Figure 2.3.4.2.4.4-1 Lunar Throttling Range Required

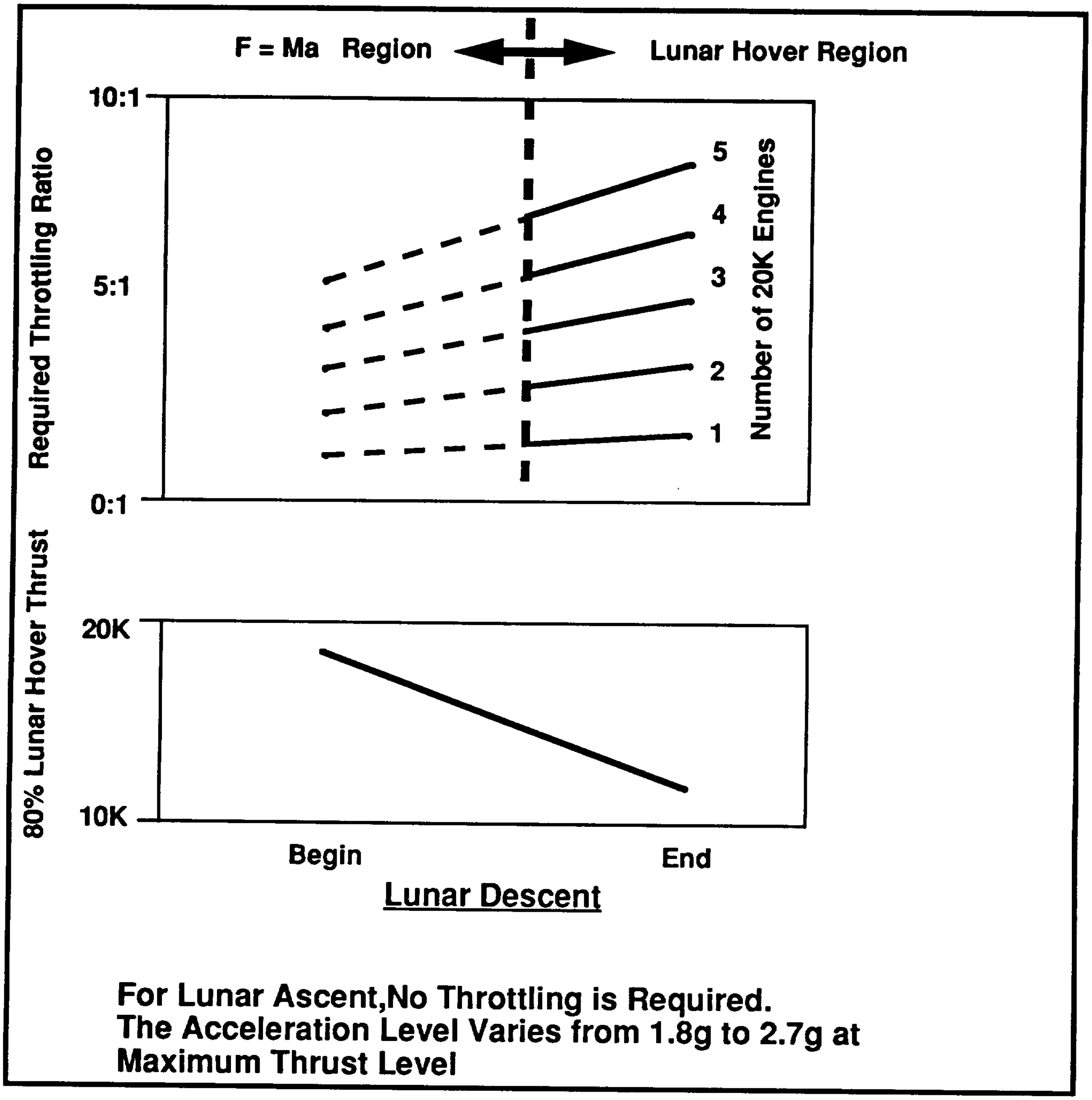


Figure 2.3.4.2.4.4-2 Throttling Ratio, Including Engine-Out (4E-5B)

ASE vs RL-10 Analysis - This analysis defines the sensitivity of ASE and the RL-10 derivatives to the performance requirements of the lunar mission. The real measures of the effectiveness of a propulsion system are only two. The first is the ability of the system to minimize initial mass in low Earth orbit (IMLEO). The second measure is the ability to place a maximum cargo mass on the lunar surface. If a new engine (ASE) can be developed which will result in a significant increase in either or both of these performance parameters, it will be worthwhile. If, however, the lack of an ASE causes the addition of even one additional ETO launch, or the addition of one additional lunar mission due to inability to transfer the required tonnage, then the development of the ASE will have been the overwhelming correct choice. Figure 2.3.4.2.4.5-1 shows the sensitivity of the engine selection to the IMLEO, for an expendable cargo mission delivering 37.4 tonnes, and for a piloted mission delivering 14.6 tonnes. The RL10A-4, RL10B-2 and ASE engines are used in making this assessment. As can be seen, the largest reduction in IMLEO due to using an RL10A-4 is approximately 8%, and if an RL10B-2 is used, the ASE only saves approximately 4% in required IMLEO. Since the performance of the HLLV is not fixed at this time, it may be more cost effective to influence its capacity, rather than to develop ASE technology.

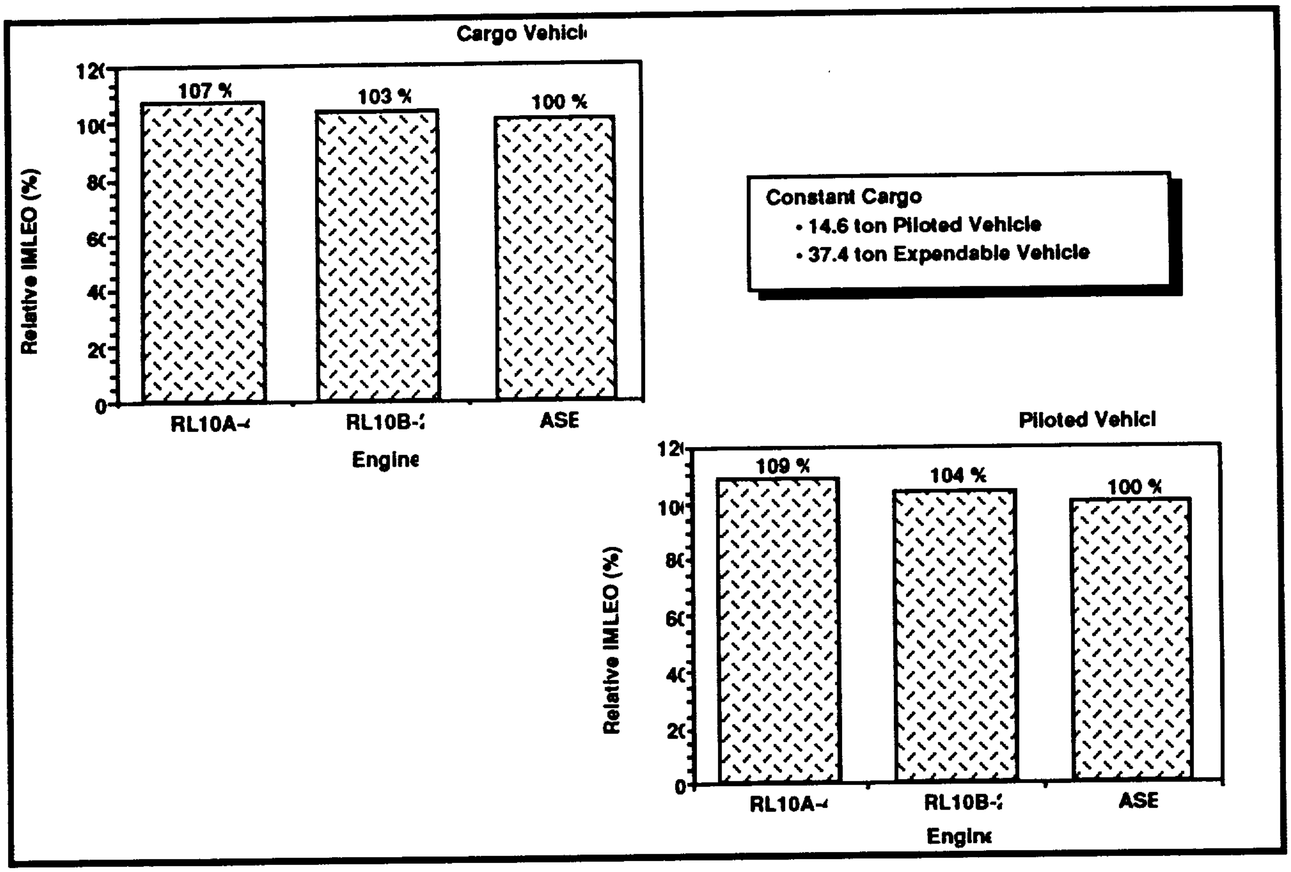


Figure 2.3.4.2.4.5-1 Relative IMLEO vs Engine Type

As was the case for IMLEO, the three engines are compared, but in this case to the quantity of tonnage delivered to the lunar surface in both an expendable cargo and a piloted mission mode, Figure 2.3.4.2.4.5-2. These calculations are made assuming a constant IMLEO, and a constant vehicle configuration. The effect of using the different engines is more pronounced than the IMLEO evaluation, but the overall effect is similar. The use of an ASE rather than an RL10A-4 provides an increased lunar cargo of approximately 50% for a piloted mission, and approximately

8% for an expendable cargo mission. This is certainly very significant, but if an RL10-B2 was available, the increase is reduced to 14% for a piloted mission, and to 3% for an expendable cargo mission.

In that the architecture for manned space exploration is somewhat flexible at this time, the best approach may be to have parallel technology paths until mission profiles are better established, using the RL-10 technology that is in hand today, modified by strategies to work space basing, ICHM, and the other engine technology issues, then evolve to ASE, IME, or even more advanced RL-10 derivatives, as appropriate to meet the "to be defined" missions.

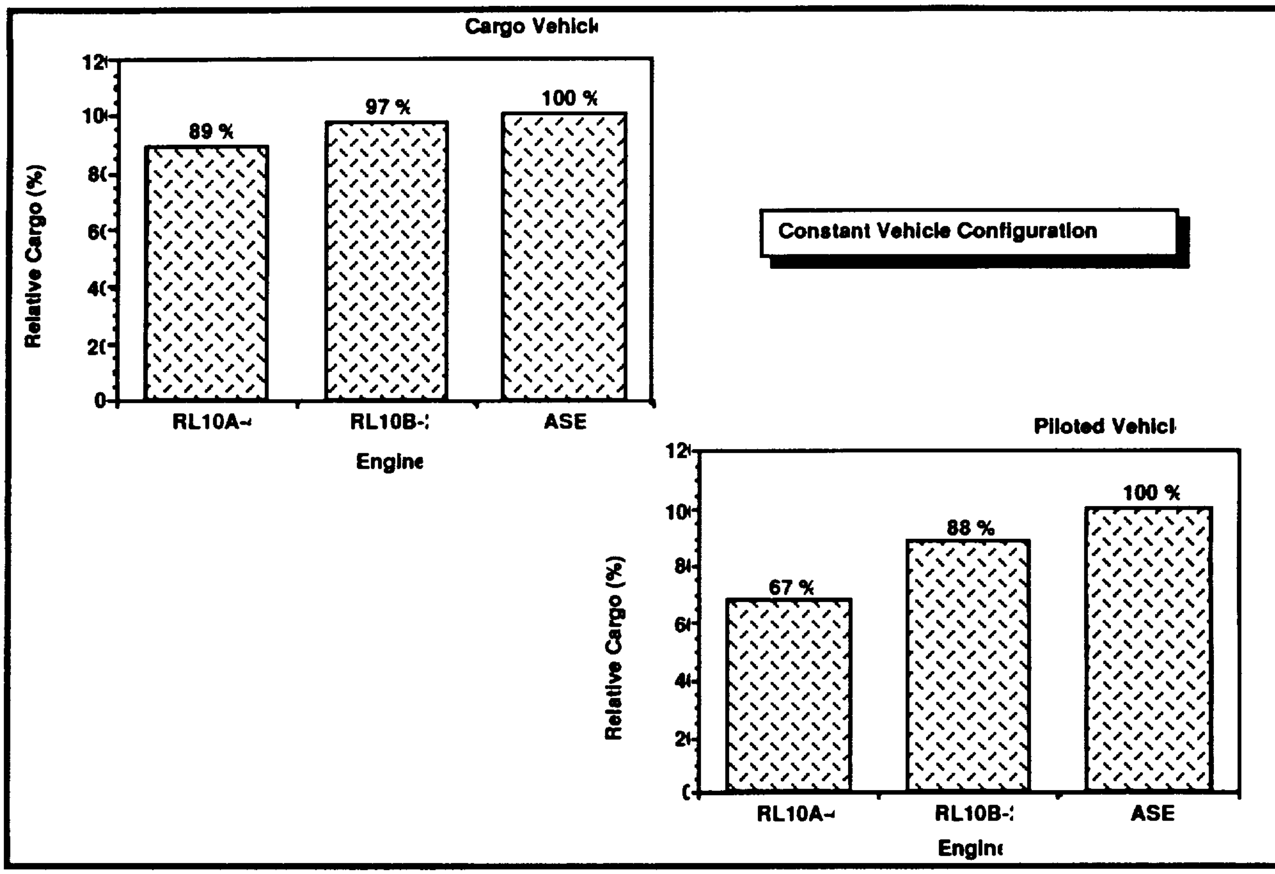


Figure 2.3.4.2.4.5-2 Relative Cargo vs Engine Type

2.3.4.2.5 Fluid Management

Minimization of Fluid Services - An evaluation was made of the fluids required to be provided to an STV vehicle system during the various phases of its operational life. This evaluation considered those fluids required at a launch facility, and at SSF or other LEO node. It considered first the initial mission of an STV that is likely to be expendable and which may not integrate all of the eventual technologies. It may also require little or no support from SSF. Next it considered that the STV is operational, but is basically a core vehicle, useful for near earth missions, or for missions that do not require drop tanks. It is considered that this vehicle will be primarily space based. Eventually, drop tanks will be incorporated, and the capability will exist for GEO missions, or for heavy cargo missions. The final STV configuration would include a crew cab, and be a manned vehicle capable of lunar missions. The application of key technologies during these phases is important in minimizing the operational complexity of the STV system. If these technologies are incorporated, the fluids that need to be supplied from SSF to support each mission consist primarily of H/O. If propellant grade H/O is converted to fuel cell grade H/O and

Table 2.3.4.2.5.1-1 Fluid Systems Support Required of ETO & SSF

Initial STV Mission	Operational STV Mission	GEO or Heavy STV Mission	Lunar Mission
ETO			
H/O Supply-Core H/O Vent GN2 Purge-Cargo Bay HP H/O-Integral RCS	H/O Supply-Core H/O Vent Gn2 Purge-Cargo Bay	H/O Supply-DropTnks H/O Vent GN2 Purge-Cargo Bay	H/O Supply-DropTnks H/O Vent GN2 Purge-Cargo Bay LN2 Supply-Breathing H2O Supply-Crew/ Shield
(Technology Driven) H/O Supply-Fuel Cells HP He-Engine HP He-Tank Pressn Hydraul Fluid-Gimb Act N2H4 Supply-RCS N2H4 Haz Vent	H/O Supply-Fuel Cells HP He-Engine HP He-TankPresn HydraulicFluid-Gimb Act N2H4 Supply-RCS N2H4 Haz Vent	H/O Supply-Fuel Cells HP He-Engine HP He-Tank Pressn Hydraulic Fluid-Gimb Act N2H4 Supply-RCS N2H4 Haz Vent	H/O Supply-Fuel Cells HP He Engine HP He-Tank Pressn Hydraulic Fluid-Gimb Act N2H4 Supply-RCS N2H4 Haz Vent
SSF (or other node)			
None	H/O Supply-Core	H/O Supply-Drop Tanks	H/O Supply-DropTnks H2O Supply-Crew/ Shield LN2 Supply-Breathing
	(Technology Driven) H/O Supply-Fuel Cells HP He-Engine HP He-Tank Pressn Hydraulic Fluid-Gimb Act N2H4 Vent-RCS	H/O Supply-Fuel Cells HP He-Engine HP He-Tank Pressn Hydraulic Fluid-Gimb Act N2H4 Vent-RCS	H/O Supply-Fuel Cells HP He-Engine HP He-Tank Pressn Hydraulic Fluid-Gimb Act N2H4 Vent-RCS

breathing grade oxygen by use of molecular sieve or similar technology, the requirement to supply these commodities is eliminated. Similarly, the pressurization system for the tankage, and the purge and valve actuation support for the engines must not require helium, or helium must be serviced. The engines must utilize a thrust vector control system, or at the minimum, electromechanical actuators, to eliminate the requirement for hydraulic fluid. One of the most important issue deals with the RCS system. The RCS system must be based on H/O in order to eliminate the complex operations dealing with storable propellants. The ultimate cost of the STV system will be strongly influenced by the cost to provide operational support. It is imperative to limit the operational fluids to the absolute minimum number. Table 2.3.4.2.5.1-1 lists the operational fluids and their servicing locations.

The preliminary fluid schematic for the core vehicle shown in Figure 2.3.4.2.5.1-1 incorporates the technologies necessary to prevent servicing of a number of fluids. The core tank pressurization system is shown to be autogenous. The RCS system is shown as it might appear using a H/O system, with gas generators providing heat and power to gasify the propellant and allow storage as a high pressure gas. Fuel cell hydrogen and oxygen, and oxygen for the crew is supplied thru

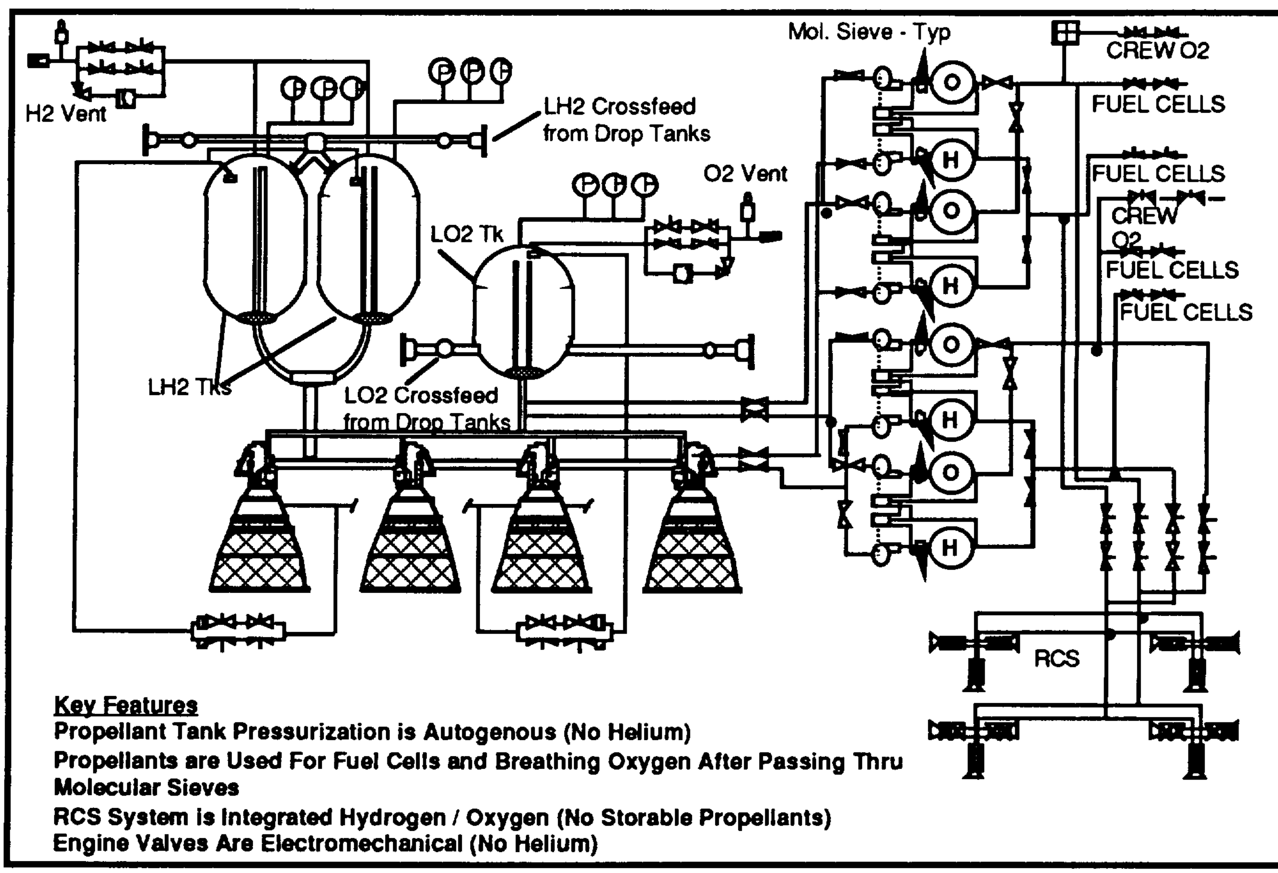


Figure 2.3.4.2.5.1-2 Core Tanks Propulsion and Fluids Schematic

molecular sieves. Engine functions do not require helium. This schematic was generated with the 90-day configuration in mind, but most of the pertinent items are applicable to the 4E-5B or other vehicles as well. This particular configuration benefited from flowing all propellant through the core tanks as a manifold. This was done to simplify the plumbing be eliminating a number of valves and simplify operation. As will be seen, the 4E-5B configuration has gone back to a more conventional manifold approach.

Propellant Settling and Transfer - Figure 2.3.4.2.5.2-1 shows the routing of the propellant lines of the 4E-5B vehicle from the drop tanks to the core vehicle and from the core vehicle to the aerobrake. It is a useful reference when evaluating force vectors to be used in the settling and transfer of propellants during each of the mission phases. Due to the size of the present core vehicle tanks, and requirement to feed and operate from the tanks in the aerobrake, the concept now includes a manifold for engine feed, as shown in the figure.

A number of methods may be used to settle the cryogenic propellants in order to accomplish transfer to another tank or to the engines. These methods have been evaluated by detailed mission phase, to assure that the configuration of the vehicle, its orientation, and the flow requirements

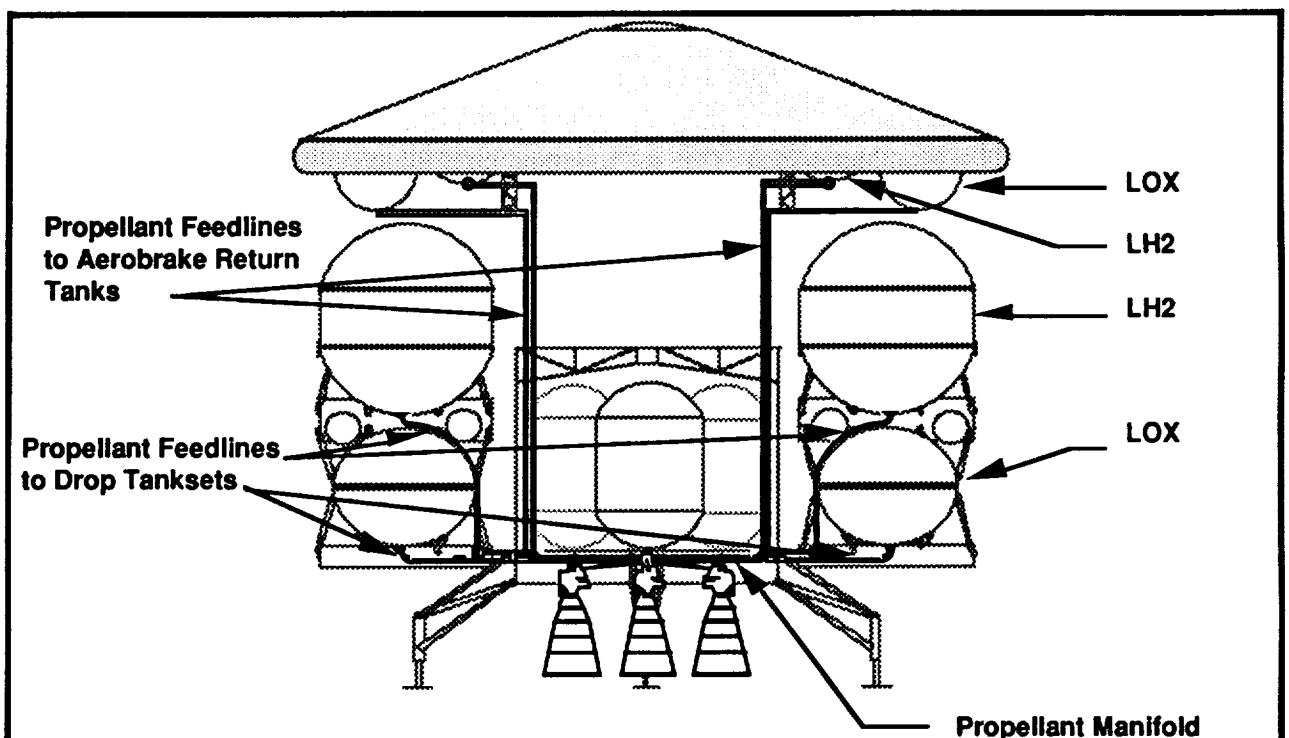


Figure 2.3.4.2.5.2-2 Propellant Feed System

have been properly considered. This evaluation only applies to the 4E-5B vehicle, although other configurations result in similar results. Table 2.3.4.2.5.2-1 presents this logic.

A study was conducted to select the optimum pressurization method for the STV LOX tanks. The study compared autogenous pressurization using gaseous oxygen against helium pressurization.

The criteria for assessing the benefits of each system include looking at cost, weight, risk/reliability, responsiveness, impact on vehicle performance and operation. These criteria were quantified as much as possible to provide a more concrete comparison.

LOX Tank Pressurization Study - Figure 2.3.4.2.5.2-1 shows the objective, criteria, groundrules and assumptions established to facilitate the study.

- **Objective:**
  - **To Select the Optimum Pressurization Method for the LOX Tankage. The Study Compares Autogenous Pressurization from the Engines to Helium Pressurization from Pressurized Storage Containers.**
- **Criteria for Assessment:**
  - **Cost**
  - **Weight**
  - **Complexity**
  - **Risk / Reliability**
  - **Responsiveness**
  - **Vehicle Performance**
  - **Operations**
- **Groundrules & Assumptions:**
  - **4E-5B Configuration**
  - **Initial Tank Pressure is 25 psia and Increased to 32 psia During Burn**
  - **5% Initial Tank Ullage**
  - **Engine Has Provisions for Autogenous LOX Pressurization and Does Not Require Helium**
  - **GOX Pressurant Temperature = 350°R**
  - **Helium Pressurant Temperature = 500°R**
  - **4 Engines with a Total Propellant Flowrate of 141.8 lb/s (64.3 kg/s)**

Figure 2.3.4.2.5.2-1 LOX Tank Pressurization Study (4E-5B)

The baselined mission for the analysis was STV configuration 4E-5B. The mission was divided into 4 phases consisting of the Trans-Lunar Injection (TLI), Lunar Orbit Injection (LOI), ascent/descent and the trans-Earth injection. Pressurization analyses were performed separately for each based on the identified tank volume and engine burn duration. These are identified in Figure 2.3.2.4.2.5.2-2. The pressurant required for the autogenous and GHe pressurized systems was

Table 2.3.4.2.5.2-1 Settling / Transfer vs Mission Phase (4E-5B)

Operation	Settling	Transfer
ETO	N/A	N/A
Mate Drop Tanks with Core Tanks Using Remote Manipulator System	N/A	N/A
Hydrogen Transfer from Drop Tanks to Core Tanks	Primary: Vent Gas Settling Secondary: RCS, Atmospheric Drag, or Resistojets	Primary: Gravity Head Secondary: Pump (Vented or Non-Vented)
Oxygen Transfer from Drop Tanks to Core Tanks	Primary: Vent Gas Settling Secondary: RCS, Atmospheric Drag, or Resistojets	Pump
Hydrogen Transfer from Drop Tanks to Aerobrake Tanks	Primary: Vent Gas Settling Secondary: RCS, Atmospheric Drag, or Resistojets	Primary: Pressure Transfer Secondary: Pump
Oxygen Transfer from Drop Tanks to Aerobrake Tanks	Primary: Vent Gas Settling Secondary: RCS, Atmospheric Drag, or Resistojets	Primary: Pressure Transfer Secondary: Pump
Trans Lunar Injection	RCS, or THI to Pump Head Idle to Full Engine	Gravity Head
Mid Course Correction	Primary: N/A (RCS Compressed Gas System) Secondary: RCS, or Tank Head Idle to Pump Head Idle to Full Engine	Primary: Gaseous Pressure Secondary: Gravity Head
Lunar Orbit Insertion	RCS, or THI to Pump Head Idle to Full Engine	Gravity Head
Aerobrake RCS in LLO	N/A (Compressed Gas)	Gaseous Pressure
Lunar Descent	RCS, or THI to Pump Head Idle to Full Engine	Gravity Head
Lunar Ascent	Lunar Gravity	Gravity Head
Trans Earth Injection	RCS, or THI to Pump Head Idle to Full Engine	Gravity Head
Mid Course Correction	Primary: N/A (RCS Compressed Gas System) Secondary: RCS or Tank Head Idle to Pump Head Idle to Full Engine	Primary: Gaseous Pressure Secondary: Gravity Head
Aeropass	N/A (RCS Compressed Gas System for Control)	Gaseous Pressure
Orbit Circularization	Primary: N/A (RCS Compressed Gas System) Secondary: RCS or Tank Head Idle	Primary: Gaseous Pressure Secondary: Gravity Head

determined analytically. The autogenous, (which uses warm GOX from the engine) and the 2 methods of helium pressurization were investigated for pressurant usage as well as when evaluating the total ullage mass remaining at the end of the burn. Helium pressurization consisted primarily of injecting pressurant into the ullage and secondarily of injecting it into the LOX in order to take advantage of the partial pressure in reducing the ullage mass. Results of the analyses reveal that the pressurant required for an autogenous system is heavier than for helium pressurization. The results are summarized in Figure 2.3.4.2.5.3-2.

Helium pressurization (where pressurant was bubbled into the LOX) was eliminated from further study. Although the helium pressurant mass was reduced with this technique, the amount of GOX used in the pressurization turned out to be greater than that of an autogenous system. Thus for an STV application, this method is not practical.

Summary of the study consists of a one to one comparison between an autogenous system against GHe pressurization. The systems were compared with respect to cost, weight, complexity, risk/reliability, responsiveness, performance and operation. The conclusion based on the groundrules used is that an autogenous system will cause a performance penalty, but that this should be more than offset by the operational benefits. See Figure 2.3.4.2.5.3-4 for a discussion of the factors that effect this conclusion.

TLI **(Drop Tanks)**	**194,413 lb (88,169 kg) of LOX** **Tank Volume = 2867 ft3 (81.2 m3)** **Engine Burn Duration = 1340 s**
LOI **(Drop Tanks)**	**36,382 lb (16,500 kg) of LOX** **Tank Volume = 549 ft 3 (15.6 m3)** **Engine Burn Duration = 245 sec**
Ascent / Descent **(Core Tanks)**	**53,734 lb (24,369 kg) of LOX** **Tank Volume = 812 ft3 (23 m 3)** **Engine Burn Duration = 1st Burn 265 sec,** **2nd Burn 98 sec**
TEI **(Core Tanks)**	**12,635 lb (5,730 kg) of LOX** **Tank Volume = 201 ft3 (5.7 m3)** **Engine Burn Duration = 1st Burn 64 sec,** **2nd Burn 14 sec**

Figure 2.3.4.2.5.3-2 Parameters of Analyzed Mission

Mission Segment	Pressurization Method: Autogenous LOX Pressurization – Auto. GOX	Autogenous LOX Pressurization – Tot Ullage	GHe Pressurization Ullage Injection – GHe	GHe Pressurization Ullage Injection – Total Ullage	GHe Pressurization Bubble Into LOX – GHe	GHe Pressurization Bubble Into LOX – Total Ullage
TLI (2 LOX Tanks)	835 lb (379.5 kg)	908 lb (412.7 kg)	72.3 lb (32.9 kg)	465 lb (211.4)	59.2 lb (26.9 kg)	1201lb (545.9 kg)
LOI (1 LOX Tank)	148 lb (67.3 kg)	167 lb (75.9 kg)	13.3 lb (6.0 kg)	68 lb (30.9 kg)	9.4 lb (4.3 kg)	232.0 lb (105.5 kg)
Ascent/Descent (1 LOX Tank)	218 lb (99.1 kg)	245 lb (111.4 kg)	20.0 lb (9.1 kg)	101 lb (45.9 kg)	14.1lb (6.4 kg)	344 lb (156.4 kg)
TEI (2 LOX Tanks)	34 lb (15.5 kg)	57.4 lb (24.9 kg)	3.5 lb (1.6 kg)	24.0 lb (10.9 kg)	3.0 lb (1.4 kg)	77 lb (35.0 kg)
Total Pressurant	1235 lb (561.4 kg) of GOX		109 lb (49.5 kg) of GHe + 410 lb (186.4 kg) of GOX		85.7 lb (39 kg) of GHe + 1629 lb (740.6 kg) of GOX	

- Eliminated GHe pressurization - bubbler method, from further consideration since it requires a greater amount of GOX than the autogenous approach and still requires 86 lb (39 kg) of GHe

Figure 2.3.4.2.5.3-3 Amount of Pressurant Required

	Advantage: Autogenous LOX	Advantage: Helium Pressurization	Comment
Cost	x		Equipment Costs May Be Similar. Operational Costs Are Higher Using Helium, Assumming Equal Maintenance
Equipment Weight	x	x	1235 lb (561.4 kg) for Autogenous against 1388 lb (631 kg) for GHe Pressurization, plus gas weight on previous chart.
Complexity	x	x	Complexity of Both Methods Are Similar Since Similar Types of Components Are Involved.
Risk/Reliability	x		Slightly Greater Risk Is Associated with Helium Pressurization Due to Higher Pressure Requirement (3500 psia vs 300 psia for Autogenous)
Responsiveness	x		Helium Loading Is Eliminated. Non-condensible GHe in Tank Complicates On-orbit Resupply
Vehicle Performance Effect		x	Lower Tank Ullage Mass Is Left for GHe Pressurization after Engine Burn - Autogenous Penalizes Vehicle To Carry an Additional 2119 lb (961 kg) of Propellant
Operations	x		Autogenous System Reduces Number of Different Fluids That the Vehicle Needs To Carry by Completely Eliminating GHe Usage and Eliminates On-orbit GHe Resupply Requirement

Figure 2.3.4.2.5.3-4 LOX Autogenous vs GHe Pressurization Summary

2.3.4.2.6 Insulation

LTV Insulation Study - A study was performed to evaluate preliminary insulation concepts for the LTV TLI and LLO drop tanks. The 90-day configuration was used as the baseline. This analysis focused on the liquid hydrogen tanks since they represent the worst case for boiloff. All analysis was performed using the Martin Marietta Cryogenic Analysis Program (MMCAP). Ground performance for various insulation configurations was determined simulating Shuttle-C cargo bay conditions that are continuously purged with gaseous nitrogen while on the launch pad. Three insulation configurations consisting of 1.3, 2.54, and 5.0 cm of closed-cell spray-on-foam-insulation were examined and the results are shown on the facing page. As expected, the boiloff rate was minimized with increasing SOFI thickness (see Figure 2.3.4.2.6.1-1). However, since the insulating performance of SOFI is marginal once on-orbit, minimization of the SOFI thickness is desirable. Based on external tank experience and a review of Shuttle Centaur requirements, a 2.54 cm of SOFI was baselined. The on-orbit boiloff was estimated for three configurations of

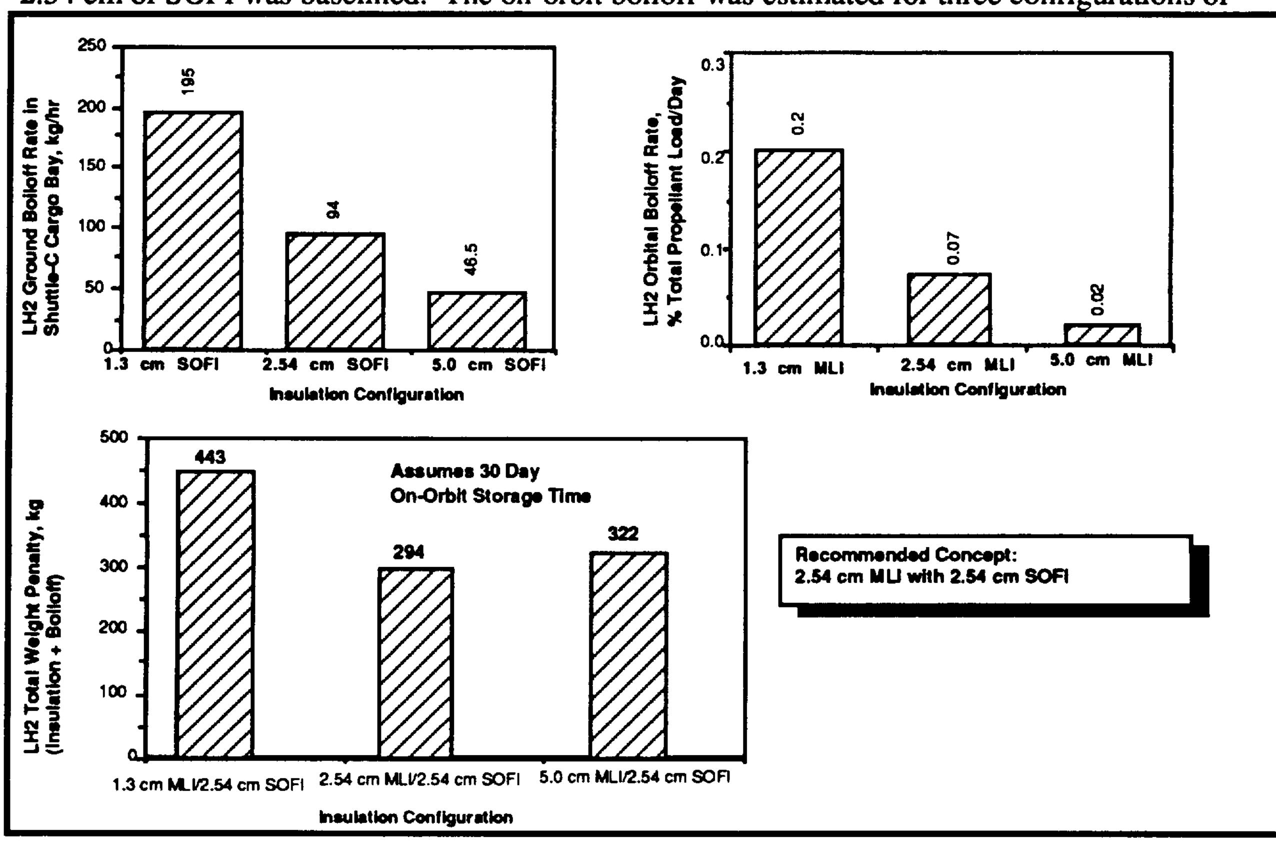


Figure 2.3.4.2.6.1-2 LTV Insulation Study Results

(MLI), 1.3 cm, 2.54 cm, 5.0 cm (with 2.54 cm of SOFI). The results are shown as percent of total propellant load per day for the TLI liquid hydrogen tanks. Since a combination of SOFI and MLI will be required on the tanks for both ground thermal control and on-orbit thermal control, the total weight penalty (insulation plus boiloff) was calculated for various SOFI/MLI combinations. Assuming a thirty day on-orbit period before the mission begins, the combination of 2.54 cm of SOFI with 2.54 cm of MLI provided the lowest weight penalty. If the on-orbit hold period is longer, however, such as sixty days, the 5.0 cm MLI with 2.54 cm of SOFI will have the lowest weight penalty.

LEV Insulation Study - A study was performed to evaluate preliminary insulation concepts for the LEV while on the lunar surface and on-orbit. Again, the liquid hydrogen tanks were examined since they are the worst case. Various thicknesses of MLI, ranging from 1.3 cm to 10 cm were evaluated for both cases. The results appear in Figure 2.3.4.2.6.2-1, and were obtained using the Martin Marietta MMCAP model. The lunar surface conditions provide the worst case thermal

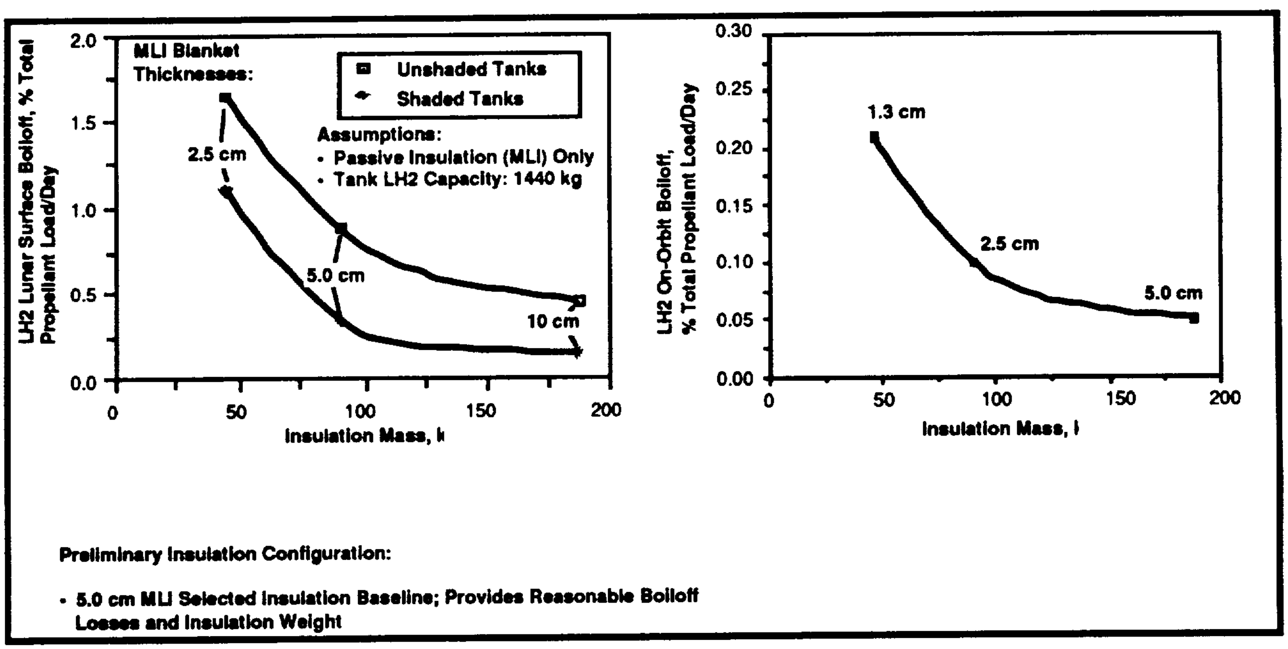


Figure 2.3.4.2.6.2-1 LEV Insulation Study Results

environment for the LEV storage tanks, particularly during the lunar day cycle. The MMCAP analysis showed that shading of the tanks during the lunar day is desirable to limit the boiloff. Only passive insulation concepts were considered in this analysis. Further reductions in boiloff could be realized if active cooling, such as mechanical refrigeration, or the addition of a vapor -cooled shield were added. On-orbit boiloff was also calculated for various MLI thicknesses. These boiloff losses were lower than on the lunar surface due to lower solar heating while on-orbit. Based on the two sets of analyses, an insulation configuration of 5.0 cm of MLI was chosen as a baseline for further study. This configuration will provide a boiloff of approximately 0.25% per day while on the lunar surface and 0.1% per day while on-orbit.

Propellant Transfer Losses - An analysis of propellant transfer under low gravity conditions from the LTV LLO tanks to the LEV liquid hydrogen and liquid oxygen tanks was performed to estimate the process efficiency for various initial conditions. The MMCAP model was again used to simulate the low-g transfer process using models developed under the Cryogenic On-orbit Liquid Depot, Storage, Acquisition, and Transfer Satellite (COLD-SAT) program by Martin Marietta. Losses occur during the transfer process from the chilldown of the tanks, plumbing, and components. The preliminary analysis accounted only for tank chilldown, however. The results are shown in Figure 2.3.4.2.6.3-1 as a percentage of the total amount transferred to the LEV. The transfer efficiency can range from near zero (for an initially wet LEV tank) to about 7% for an initially warm, dry LEV tank. Therefore, refueling of the LEV tanks before they can significantly warm up will greatly reduce the amount of losses incurred by the transfer process.

- **LTV To LEV Propellant Transfer Process Modelled Using Martin Marietta Cryogenic Analysis Program (MMCAP) To Determine Transfer Efficiency For Various Initial Conditions**
- **Efficiency Most Impacted By Receiver Tank Initial Conditions**

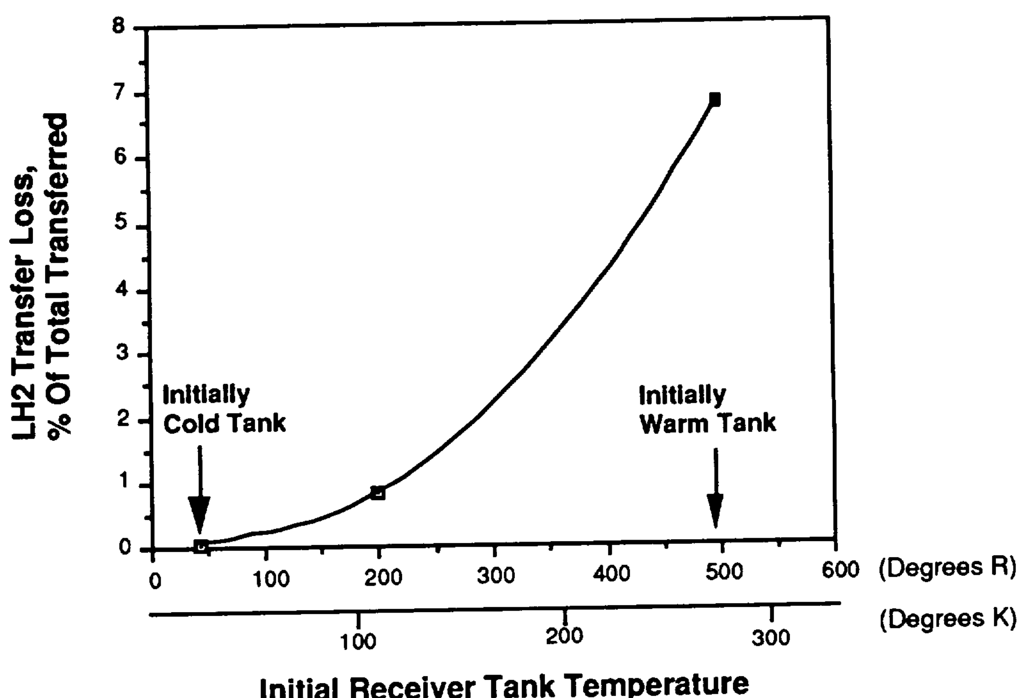


Figure 2.3.4.2.6.3-1 LTV/ LEV Propellant Transfer Analysis

MLI/Boiloff Weight Parametrics - A study was performed to develop parametric weight and performance curves for different insulation options for tank sizes typical of the LTV and LEV. The purpose of the study was to characterize the effects of tank size, on-orbit storage time, and insulation configuration of the boiloff and insulation weight. For the initial analysis, passive insulation techniques such as spray-on-foam and Multilayer Insulation (MLI) were modeled. The liquid hydrogen tanks were analyzed since they represent the worst thermal control case. Tank volumes typical of the sizes being studied for the LTV and LEV were considered (500 ft3 to 2000 ft3). Also varied was the on-orbit storage times (2 to 6 months). The MLI was modeled using test data correlations from NASA and Martin test programs. These correlations give the effective thermal conductivity of MLI "as installed" and includes the effects of seams, penetrations, and other installation imperfections that can drastically impact the overall MLI performance. In addition, the heat leak through the tank supports was included assuming the supports were made of low conductivity materials such as S-glass epoxy to limit heat leak. Boiloff weights and MLI weights were calculated varying the tank volume, storage time, and MLI thickness.

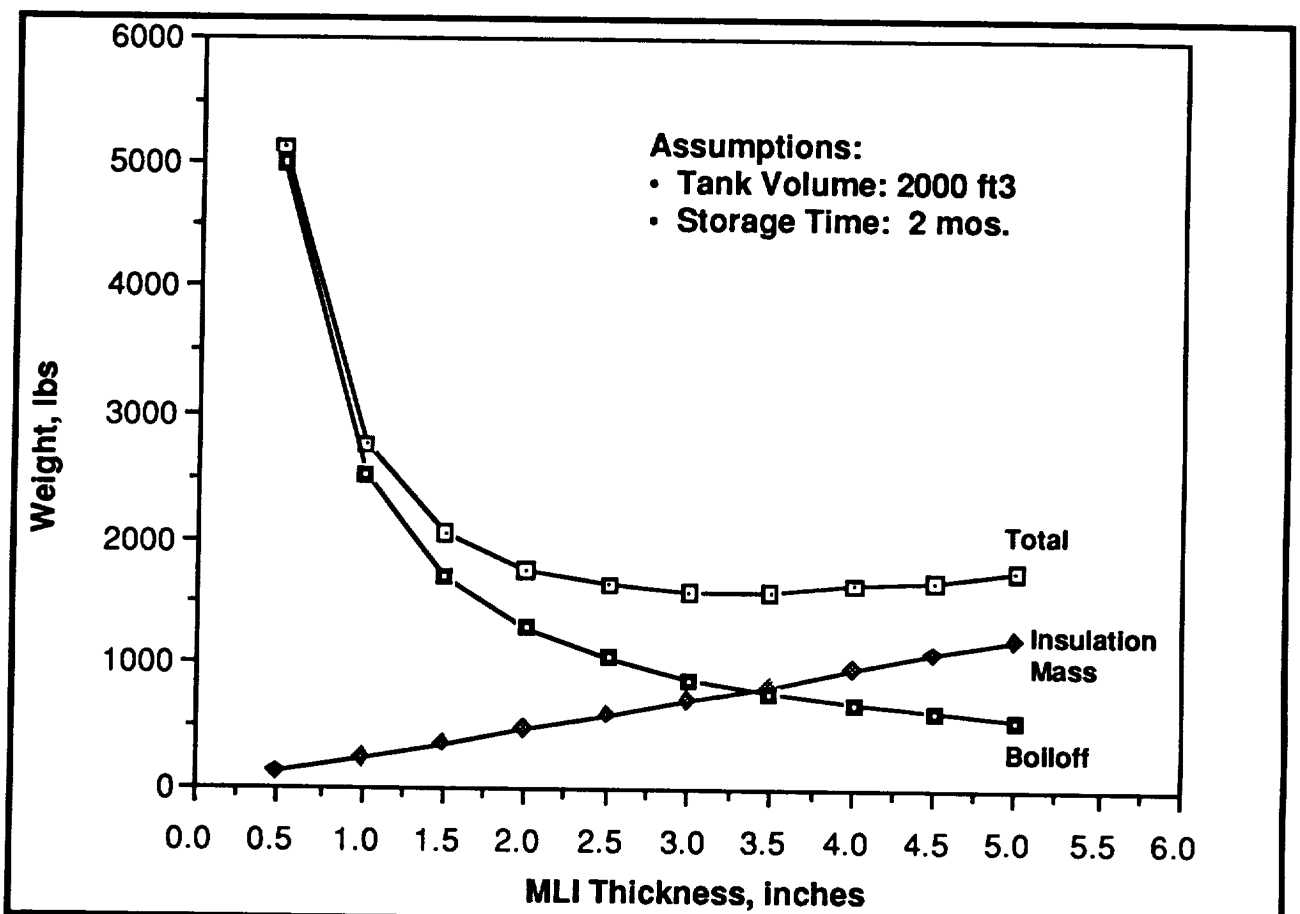


Figure 2.3.4.2.6.4-1 MLI/Boiloff Weight Parametrics

A sample of the analysis output is shown in Figure 2.3.4.2.6.4-1 for a 2000 ft3 storage tank and a two month on-orbit storage time. As shown, boiloff decreases with increasing MLI thickness while the insulation mass increases. The sum of the insulation and boiloff weights produces a curve that produces a minimum value, indicating an optimum insulation thickness for a given set of requirements. For the case shown, the optimum MLI thickness is approximately 3.0 inches.

Effect of Orbital Storage Time on Boiloff - A summary of the effect of on-orbit storage time is shown in Figure 2.3.4.2.6.5-1 for a 2000 ft3 liquid hydrogen tank. As seen, the MLI thickness is less critical for the shorter storage times. However, as the on-orbit storage time increases, the weight penalty is much more strongly effected by insulation thickness.

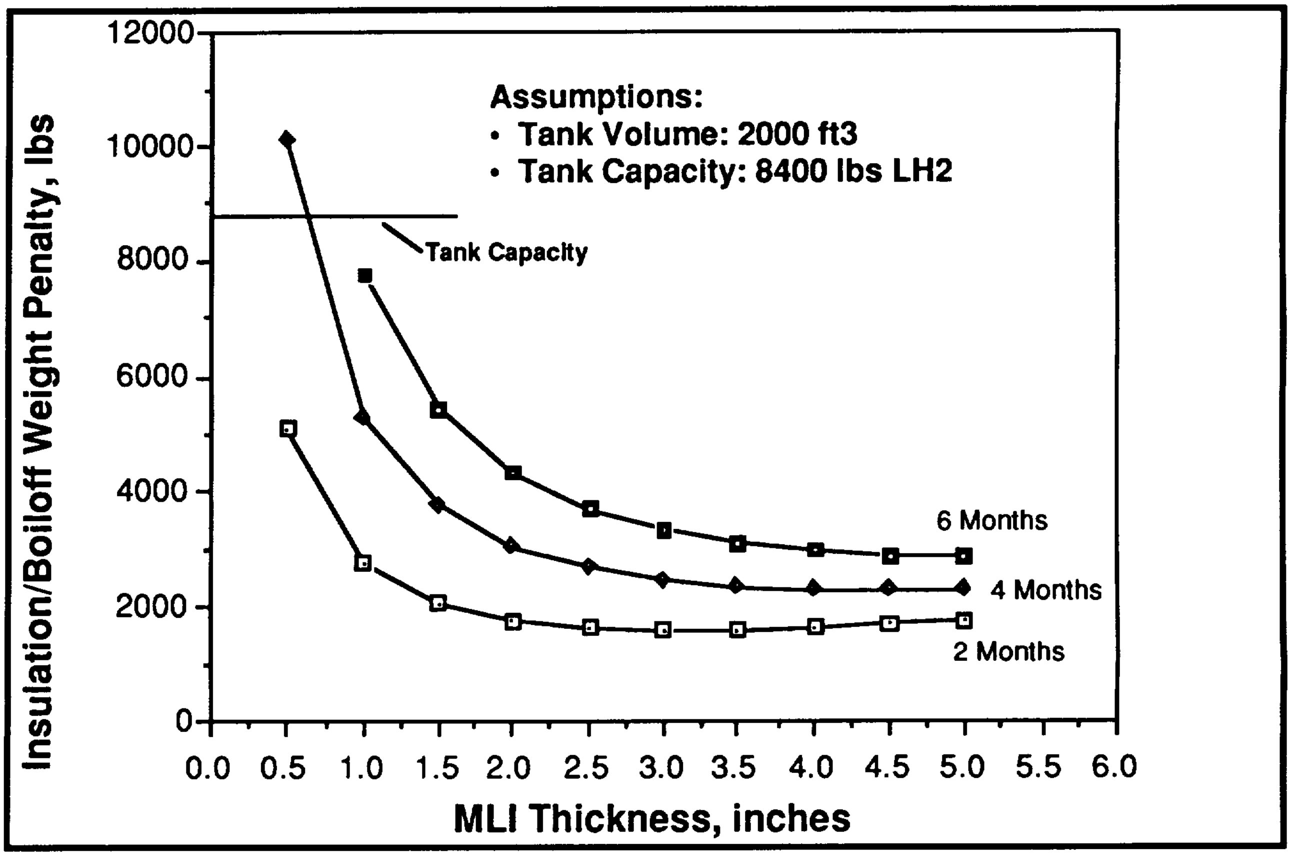


Figure 2.3.4.2.6.5-1 Effect of LEO/LLO Storage Time on Boiloff

Effect of Tank Size on Boiloff - The effect of storage tank size is shown in Figure 2.3.4.2.6.6-1 assuming a four month orbital storage time. Also shown in the vertical axis are the individual tank capacities indicating where the entire tank contents will boil away in the four month period for a given MLI thickness. For the range of tank sizes shown, increasing the MLI thickness beyond approximately 2.5 inches does not dramatically reduce the boiloff/weight penalty.

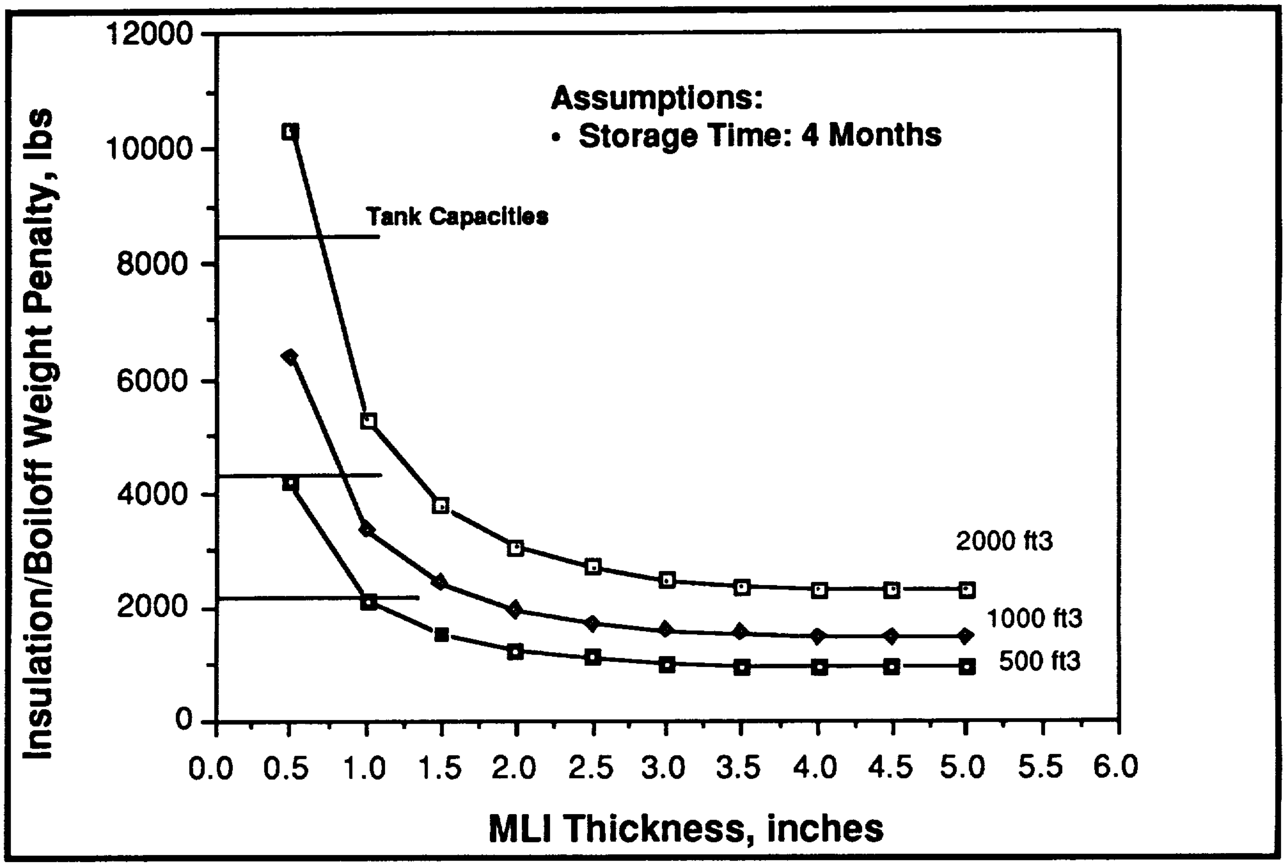


Figure 2.3.4.2.6.6-1 Effect of Tank Size on LEO/LLO Boiloff

Several conclusions may be determined from this study. First, on-orbit storage time is the major factor which impacts tank thermal insulation configuration. Second, for a long duration mission, insulation optimization is very important. Using only conventional MLI, approximately 1/3 of the propellant will be lost during a 6 month storage period. Third, use of active insulation strategies will likely be needed for long duration missions.

2.3.4.2.7 Reaction Control System

Sizing - The objective of this analysis was to define the RCS thrust levels required for the LTS. Since the detailed flight dynamics are far from definition for these vehicles, the approximate sizing of thrust levels was done in comparison to other similar vehicles. Other vehicles that have similar requirements are shown in Table 2.3.4.2.7.1-1. The Shuttle has an RCS system with two thrust levels, 25 lbf and 870 lbf. These thrusters are used at different times in a mission for different functions. The most positive control is achieved by the larger thrusters, and the fine control by the small thrusters. The acceleration levels achieved by each thruster set is shown, assuming that a minimum of two thrusters are fired at any one time. The Apollo system is also shown, which may be more representative of the STV requirements. In general, an acceleration of approximately 0.1 to 0.2 ft/sec2 is required for most attitude control and docking maneuvers, while a much higher acceleration (approximately 0.28 to 0.46 ft/sec2) is needed for aeropass control. The very small acceleration of 0.008 shown for the Shuttle is more representative of that needed for a science platform, and may be below the requirement for STV. It is also appropriate that the STV have a reasonably high acceleration level in the vicinity of SSF, to assure meeting the proximity rules.

Table 2.3.4.2.7.1-1 Historical RCS Thruster Levels vs Acceleration

Vehicle	Function	Thrust Ea. (lb)	Min. No Fired	Vehicle Wt.(lb)	Accel.(ft/sec2)
Shuttle	Docking and Attitude Contr.	25	2	200,000	0.0081
	Attitude Contr.	870	2	200,000	0.28
	Reentry	870	2	200,000	0.28
Apollo Command Module	A/B, Docking, Attitude Contr.	100	2	14,000	0.46
Apollo Service Module	Docking, Attitude Contr.	100	2	30,000	0.21
Apollo Mated Command and Service Module	Attitude Contr.	100	2	44,000	0.15

Preliminary Indications

For Attitude Control, Approx. 0.008 to 0.28 ft/sec2 (0.1 to 0.2 Nominal) is Required

For Aeropass Control, Approx. 0.28 to 0.46 ft/sec2 is Required

In order to determine the RCS thrust level for the LTS, the system was sequenced through its various mission phases to evaluate its weight and apply the appropriate acceleration level. The initial vehicle is very heavy compared to its final weight, so a considerable range of RCS thrust is expected. Figures 2.3.4.2.7.1-1, 2.3.4.2.7.1-2, and 2.3.4.2.7.1-3 show the vehicle from its initial assembly in LEO through its separation from the aerobrake in LLO. Subsequent charts show the remainder of a lunar mission sequence. The calculated thrust level requirements vary significantly, as shown.

If the core vehicle and the aerobrake are considered as a single assembly, the thrust levels required vs. mission phase as shown in Figure 2.3.4.2.7.1-4. The variation in required thrust level is extreme, ranging from approximately 100 to 1500 lbf.

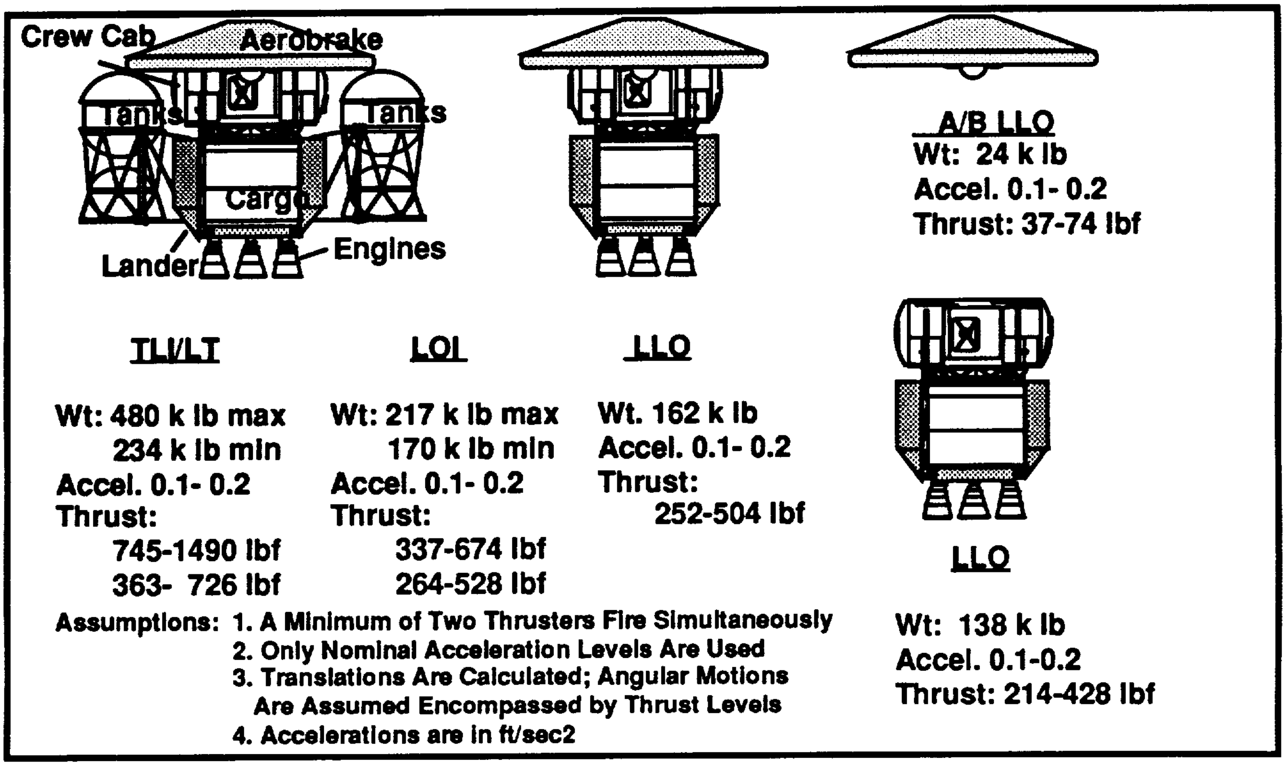


Figure 2.3.4.2.7.1-1 RCS Engine Thrust vs Mission Phase - Outbound

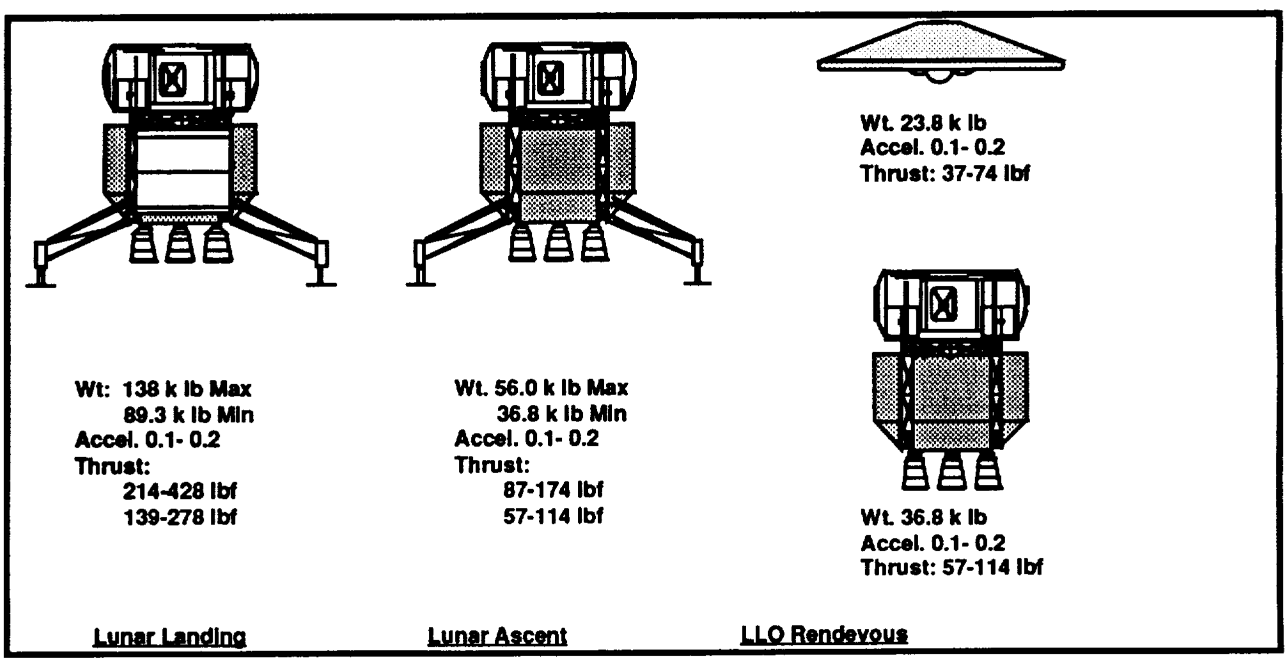


Figure 2.3.4.2.7.1-2 RCS Engine Thrust vs Mission Phase - Lunar Operations

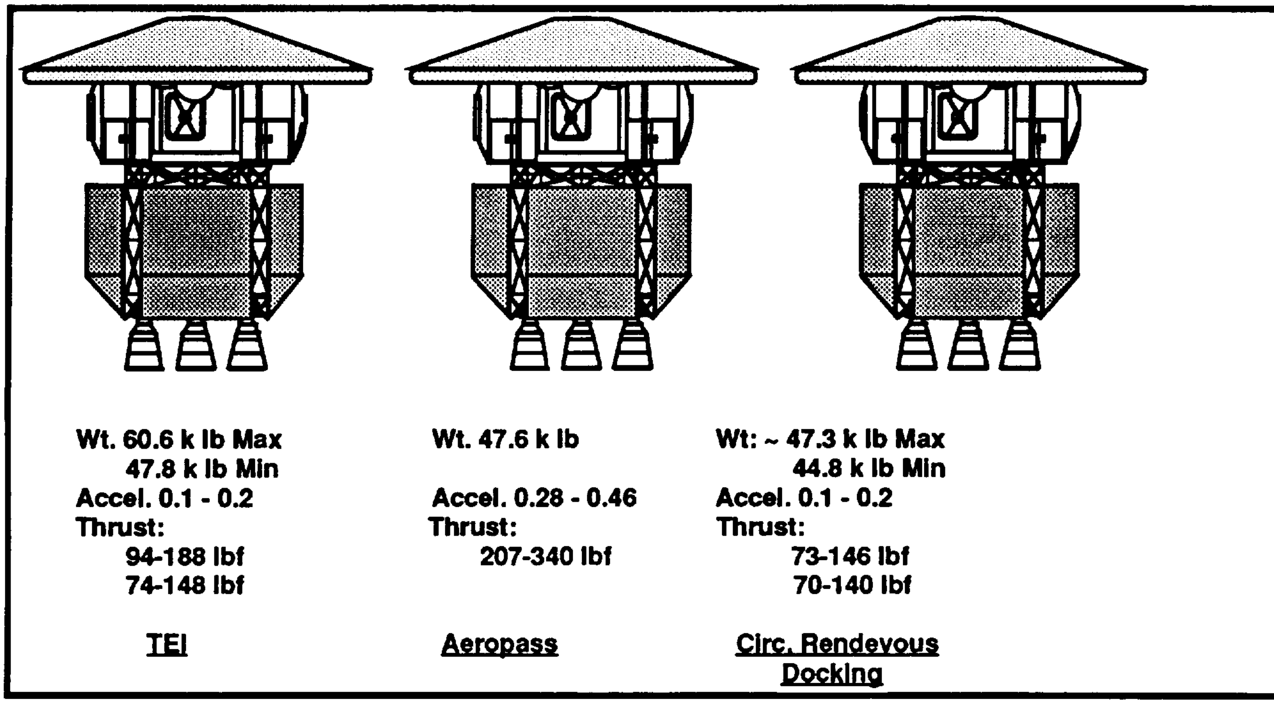


Figure 2.3.4.2.7.1-3 RCS Engine Thrust vs Mission Phase - Inbound

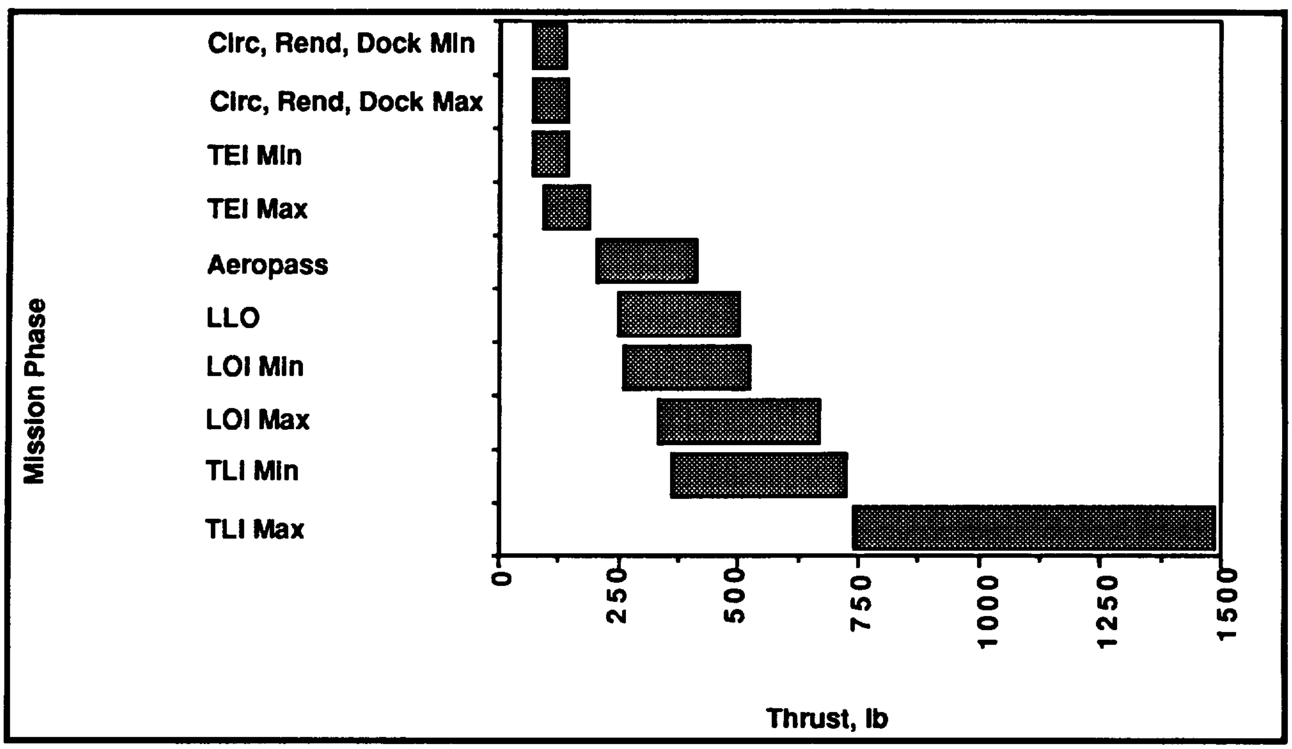


Figure 2.3.4.2.7.1-4 RCS Thrust Ranges - Core Vehicle With Aerobrake

Similarly, if the core vehicle is considered as a single assembly, the thrust levels required vs. mission phase as shown in Figure 2.3.4.2.7.1-5. The variation in required thrust level is less severe, ranging from approximately 60 to 430 lbf.

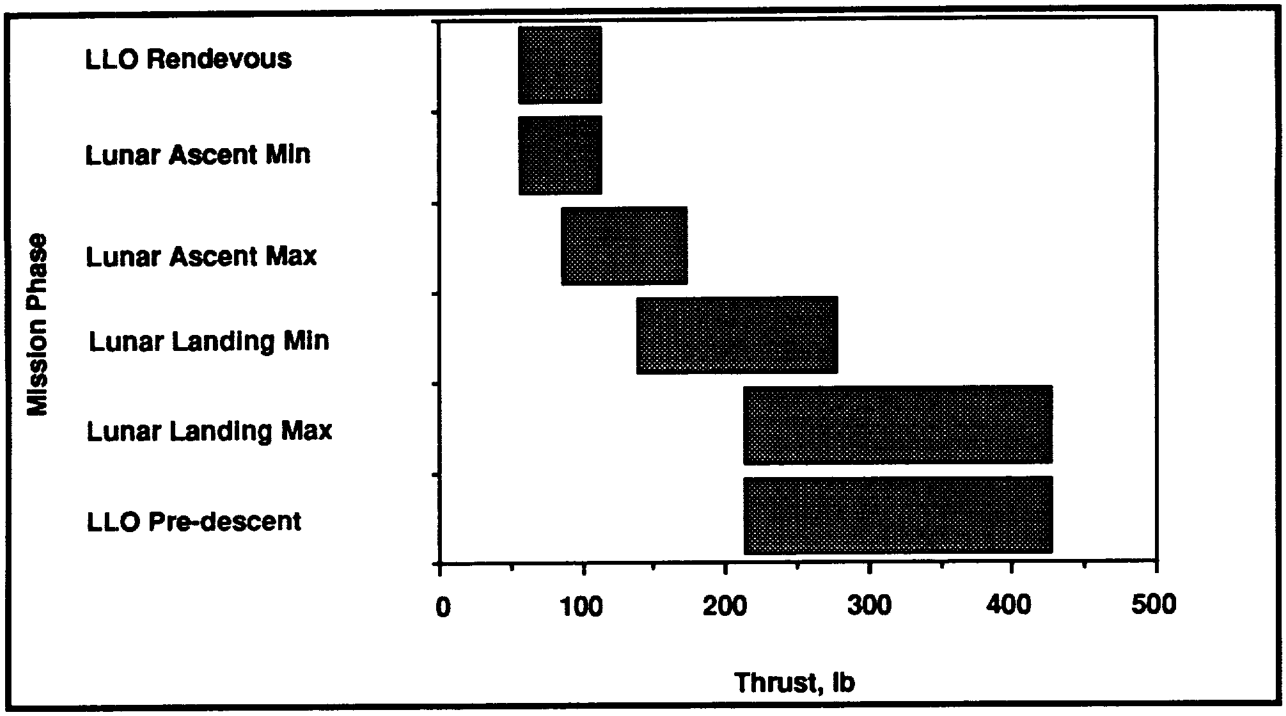


Figure 2.3.4.2.7.1-5 RCS Thrust Ranges - Core Vehicle

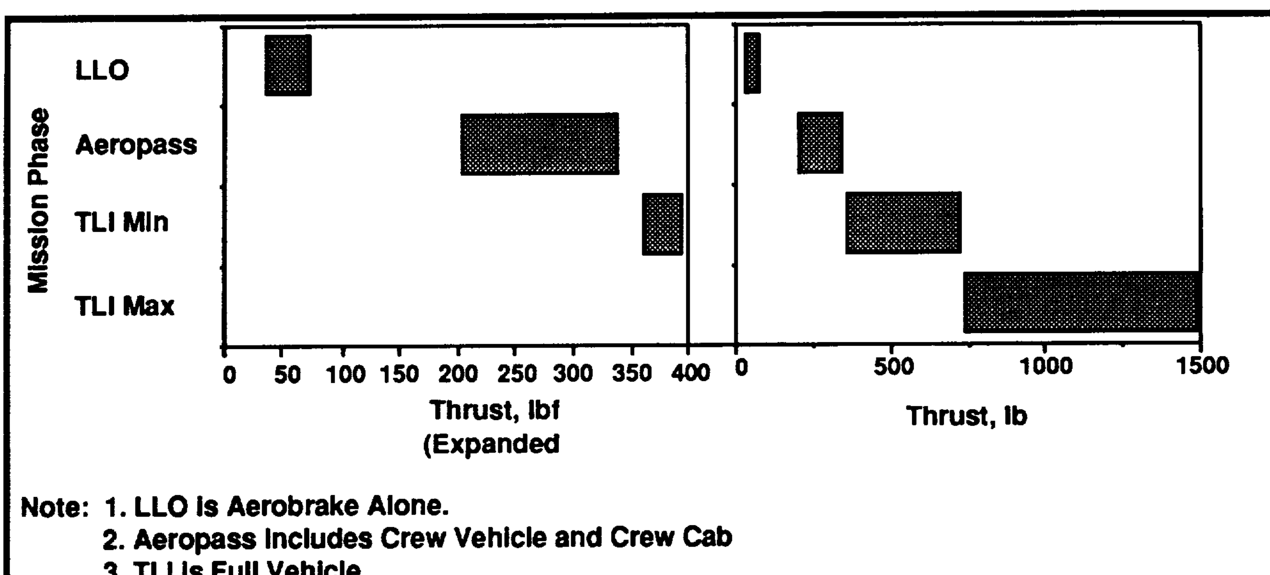


Figure 2.3.4.2.7.1-6: RCS Thrust Ranges - Aerobrake

Similarly, if the aerobrake is considered as a single assembly, the thrust levels required vs. mission phase as shown in Figure 2.3.4.2.7.1-6. In this case, all usage of the thrusters to be mounted on the aerobrake is shown, from TLI (when the full vehicle is involved) through LLO (when the aerobrake is a separate entity). The variation in required thrust level is from approximately 40 to 1500 lbf.

Selected RCS Thruster Layout - In order to avoid multiple RCS systems, it is recommended that variable thrusters be developed that will produce from approximately 50 to 1000 lbf of thrust. The preliminary arrangement of thrusters is shown in Figure 2.3.4.2.7.2-1. There are three stations that have thrusters, the top and bottom of the core vehicle, and the aerobrake. It was elected to keep thrusters off of the cargo or crew cab, even though more leverage might be applied there through a longer moment arm, in favor of a system that is self contained within the core vehicle. When the vehicle is fully assembled for a lunar mission, the thrusters at the upper end of the core vehicle are inactive. Variable thrusters are used in all locations, and 24 thruster are shown in order to provide six degrees of freedom for each assembled vehicle.

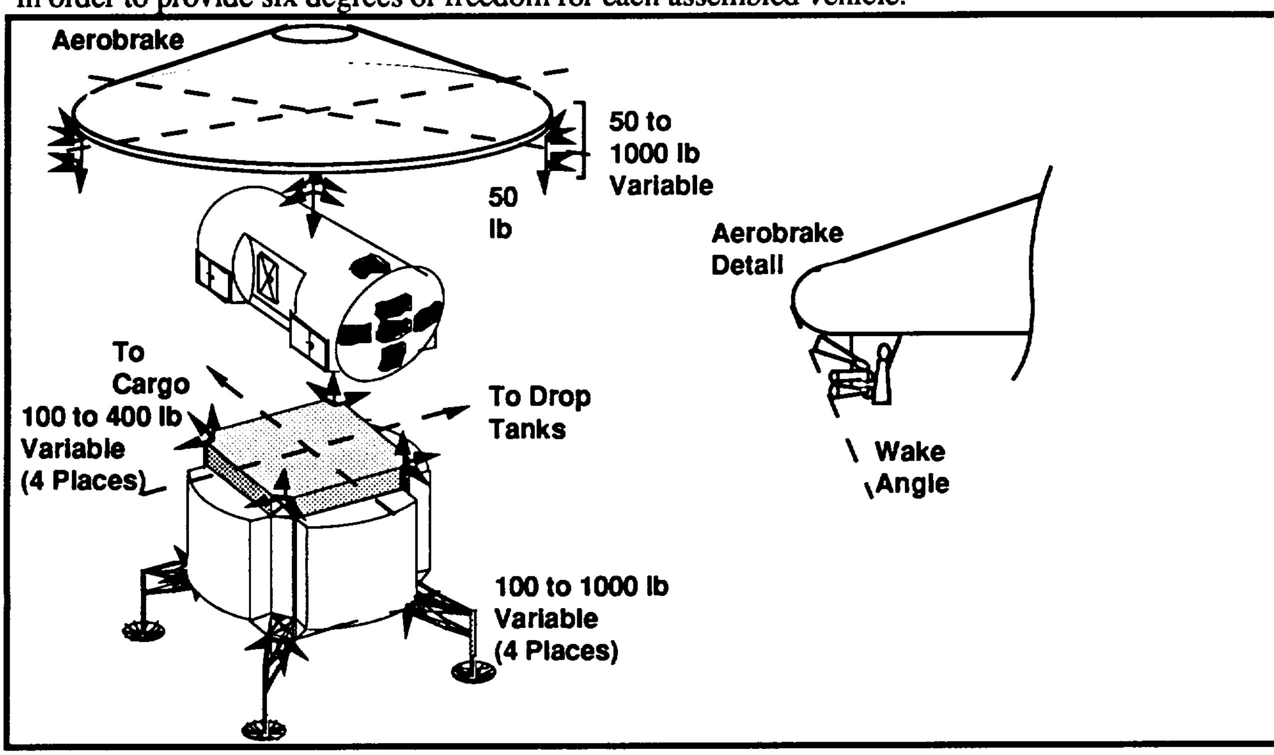


Figure 2.3.4.2.7.2-1 RCS Thruster Preliminary Arrangement

RCS System Options - The RCS system options that were considered are presented in Table 2.3.4.2.7.3-1, with some listing of the pros and cons of each type of system. The system that was selected is the bipropellant gaseous H/O, which has an inherent complexity, but provides for flowrates adequate for emergency return from the lunar vicinity, and provides a convenient gaseous supply for non-RCS functions such as crew breathing oxygen, fuel cells, and propellant tank contingency pre-pressurization.

Detail of Integrated RCS System - A schematic of the proposed integrated gaseous H/O RCS supply system is shown in Figure 2.3.4.2.7.4-1. This system resides in the core vehicle, and is supplied from the core vehicle tankage. Liquid cryogen from the core vehicle tanks is pumped to storage pressure (approximately 3500 psig) by pumps operated by gas turbines which burn gaseous hydrogen and oxygen and use their exhaust heat to gasify the fluid for storage. The

Table 2.3.4.2.7.3-1 RCS System Options

Options if Ground Based	Options If Space Based	Advantages	Disadvantages
BiPropellant MonoPropellant	Eliminated for Space Based Option Due to Storable Fluid Servicing Complexities (Non-Integrated)		
Cold Gas	Cold Gas	Potentially Simpler System	Low Performance High Pressure Storage Low Density Storage
Bi Gas (H/O)	Bi Gas (H/O)	Emerg. Return to EO Useage Flexibility High Flowrate Good Long term Stg	High Pressure Storage Low Density Storage Complex System
Cryo (H/O)	Cryo (H/O)	Low Pressure Storage High Density Storage High Isp	Large Thermal Losses Poor Long Term Storage Complex System
Supercritical (H/O)	Supercritical (H/O)	Low Pressure Storage High Density Storage	Low Flowrate (Prepres or High Demand RCS) Poor Long Term Storage Complex System

tanks are to be resupplied each time the vehicle engines are operated, which provides the necessary settling thrust for flowing liquid to the pumps. The storage tanks are sized for those intervals. This system is appropriate when using RL- 10 derivative engines, ASE engines, or the Integrated Modular Engine approach proposed by Martin Marietta and Aerojet.

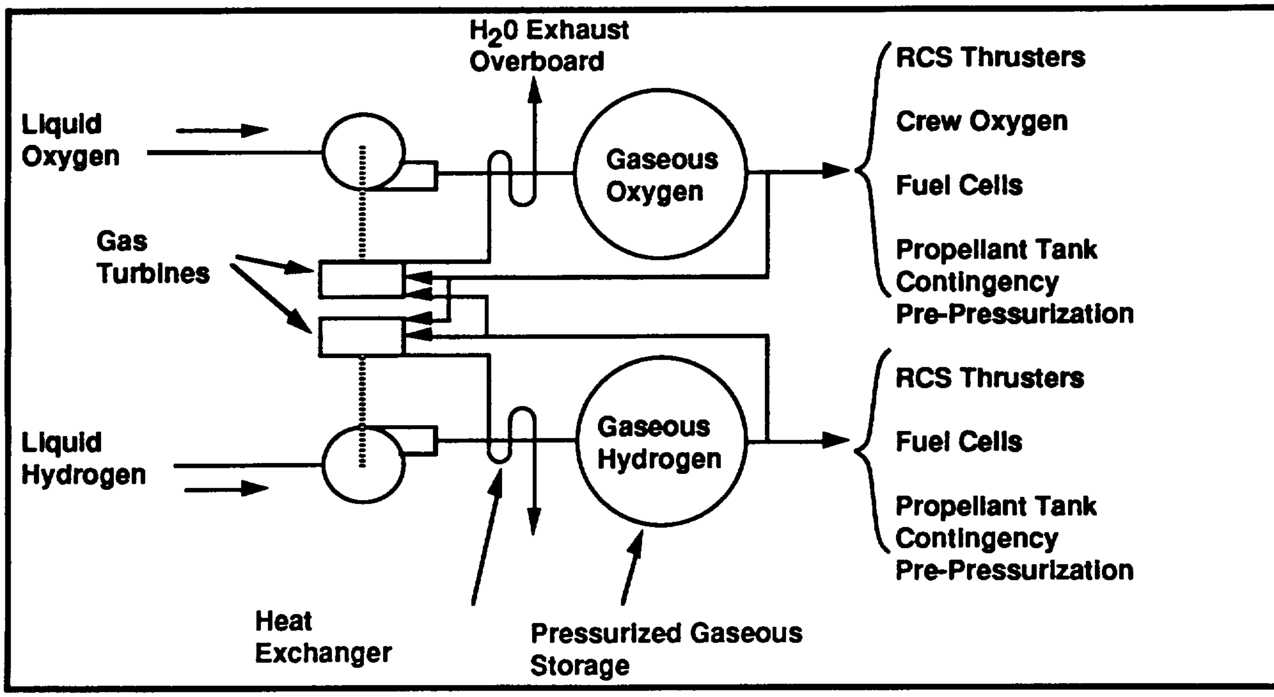


Figure 2.3.4.2.7.4-1 Integrated RCS System - Gaseous H/O

2.3.4.3 Aerobrake Analysis - The aerobrake analysis and study activity conducted as part of the STV study program provided an assessment of the benefits and issues associated with the two brake configurations - rigid and flexible - and reexamined some initial aspects of guidance and control of the system during the aeroassist maneuver.

2.3.4.3.1 Rigid Versus Flexible Trade Study - The objectives of this study were to establish the relative performance, operations and other characteristics of an all rigid versus a combination rigid/flexible aerobrake system. The key issues addressed during this study were:

- Weight
- Launch vehicle stowage (ETO)
- On-orbit assembly (or deployment) and inspection
- Entry heating
- TPS material thermal/structural capability

The groundrules and assumptions defined for this study included physical and functional characteristics and operational criteria.:

- Rigid and Flexible
    - L/D = 0.15

- W/C_DA - 14 lb/ft^2
- Blunted 70° Cone
- 24 foot maximum rigid center diameter
- Five reuses
- Robotics assembly (EVA backup)

-Rigid

FRCI 12 tile TPS material

-Flexible

- TABI flexible TPS material outboard of FRCI 12 rigid center section

- Criteria
  - Reliability
  - Weight
  - Cost
  - Operations complexity
  - Launch vehicle stowage
  - Technology development required
  - Evolution

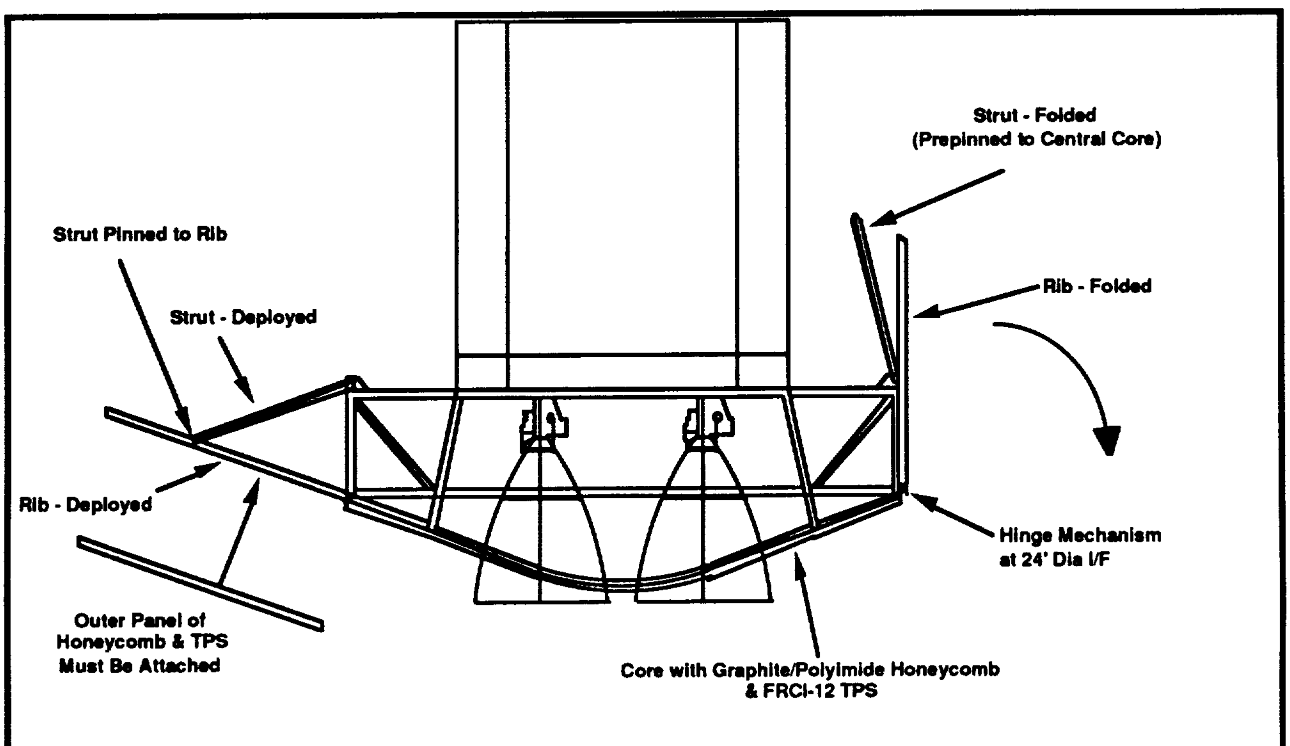


Figure 2.3.4.3.1.1-1 Eight Panel Rigid Folding Rib Aerobrake

2.3.4.3.1.1 ***Analysis***—The evaluation of the rigid aerobrake was conducted with four different configurations, an eight panel segmented design with ribs either integral to the panels or with folding ribs hinged to the center section, and a three piece design with the pieces either separate or hinged together. The eight panel rigid aerobrake with folding ribs is shown in Figure 2.3.4.3.1.1-1. It is similar to the integral rib rigid aerobrake configuration; the major difference is that the ribs (and struts) are deployable rather than being built into the panel assemblies. In the folded rib concept, the eight panel segments are attached to the ribs after the ribs are deployed by the robotics arm. Rotating the struts and pinning them to the ribs completes the LEO assembly.

The eight panel integral rib rigid aerobrake is shown in Figure 2.3.4.3.1.1-2. A space station mounted robotics arm installs each of the eight segments and their supporting struts to the 24 ft. diameter rigid center section as the center section with core module attached is rotated into position. Latching devices are engaged in mating surfaces.

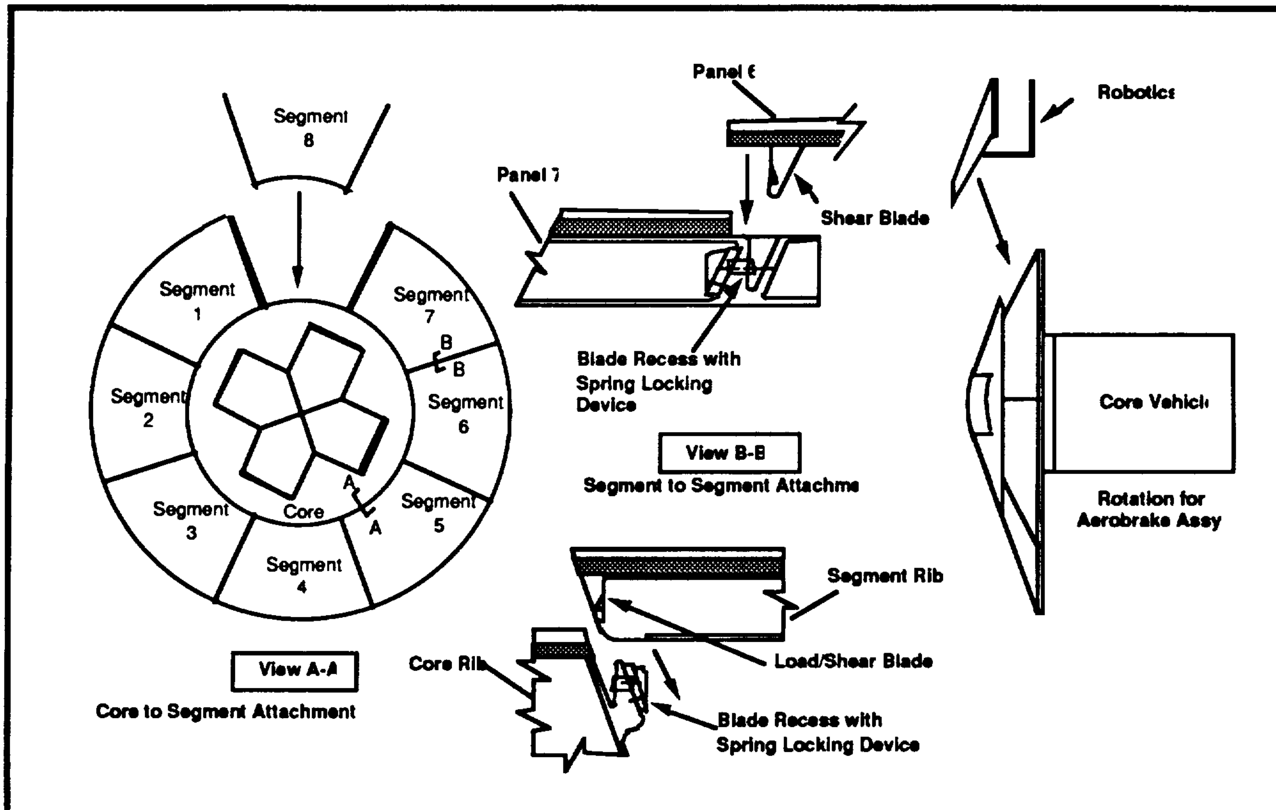


Figure 2.3.4.3.1.1-2 Eight Segment Rigid Aerobrake

The two versions of the three section rigid aerobrake are shown in Figure 2.3.4.3.1.1-3. In the hinged version, shown on the left side, the folding outboard segments require tip clearance between the tips in the folded position. Therefore, the diameter of the cargo bay of the launch vehicle must be somewhat larger than for the non-hinged version shown on the right side of the figure, since the outboard segments of that version can be fitted into the cargo bay separately.

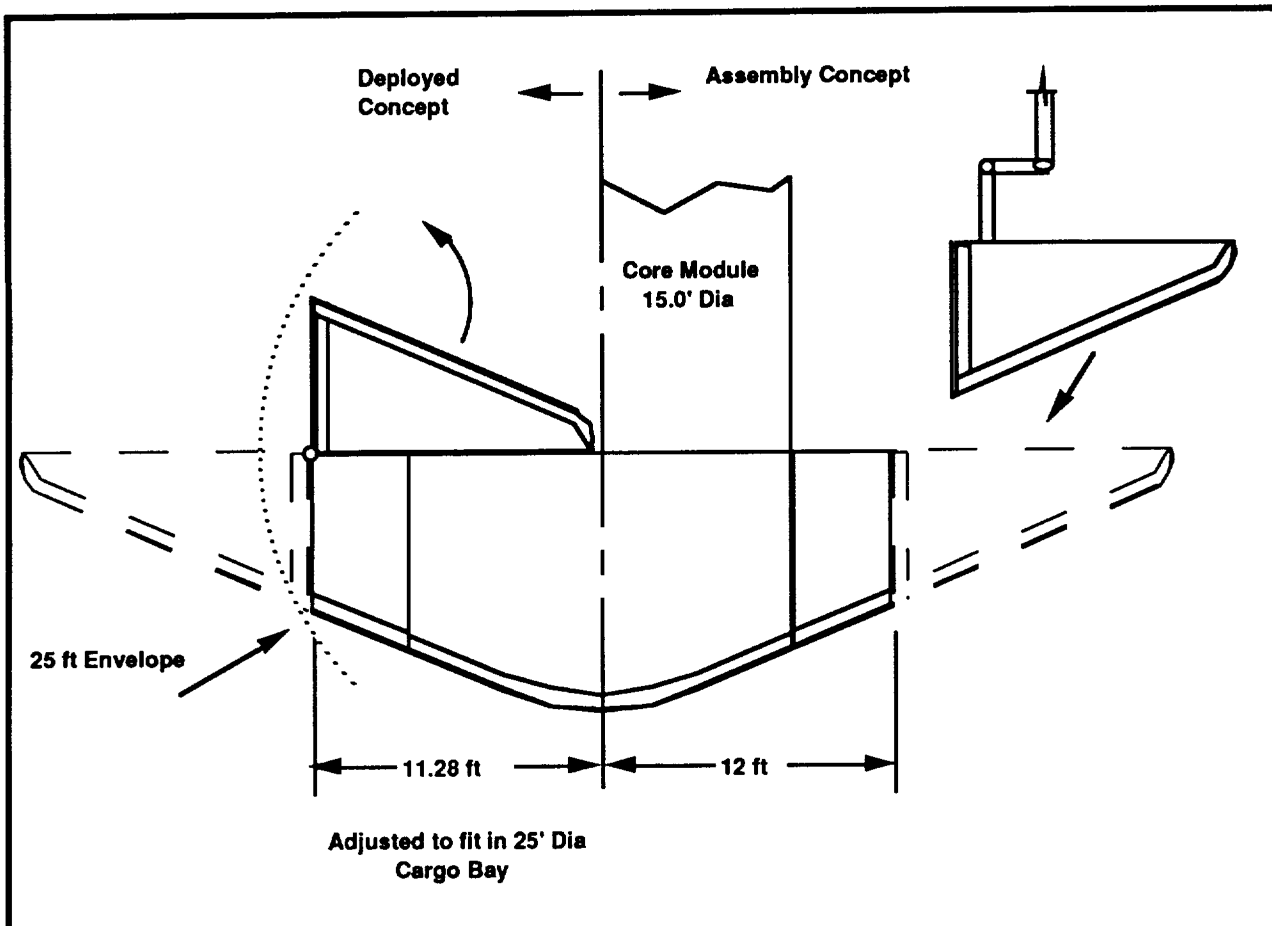


Figure 2.3.4.3.1.1-3 Three Section Deployed/Assembled (Hinged) Aerobrake

For either configuration, the 45 foot diameter aerobrake can be accommodated in a 25 foot diameter cargo bay but the core vehicle is not attached to the aerobrake during Earth launch to orbit operations.

The flexible aerobrake concept shown in Figure 2.3.4.3.1.1-4, contains 16 ribs covered by Tailorable Advanced Blanket Insulation (TABI) material outboard of the 24 ft. diameter tile protected rigid center section. A single hinged strut braces each rib. The TABI is permanently attached to the center section where it adjoins the rigid TPS material. An aerobrake diameter of 45 ft. is compatible with packaging in a 25 ft. diameter cargo bay launch vehicle although aeroheating levels and TABI temperature limits may require a larger aerobrake diameter (lower ballistic coefficient) to keep the TABI within its temperature limits. For this configuration it is desirable not to have the core vehicle attached to the aerobrake during launch to orbit, since doing so constrains the core diameter and makes the on-orbit deployment by an RMS more difficult. In any case, the hinged joint at the base of each rib must be capable of outward translation after or during rotation in order to accommodate the greater radial length the TABI material occupies in the stowed configuration.

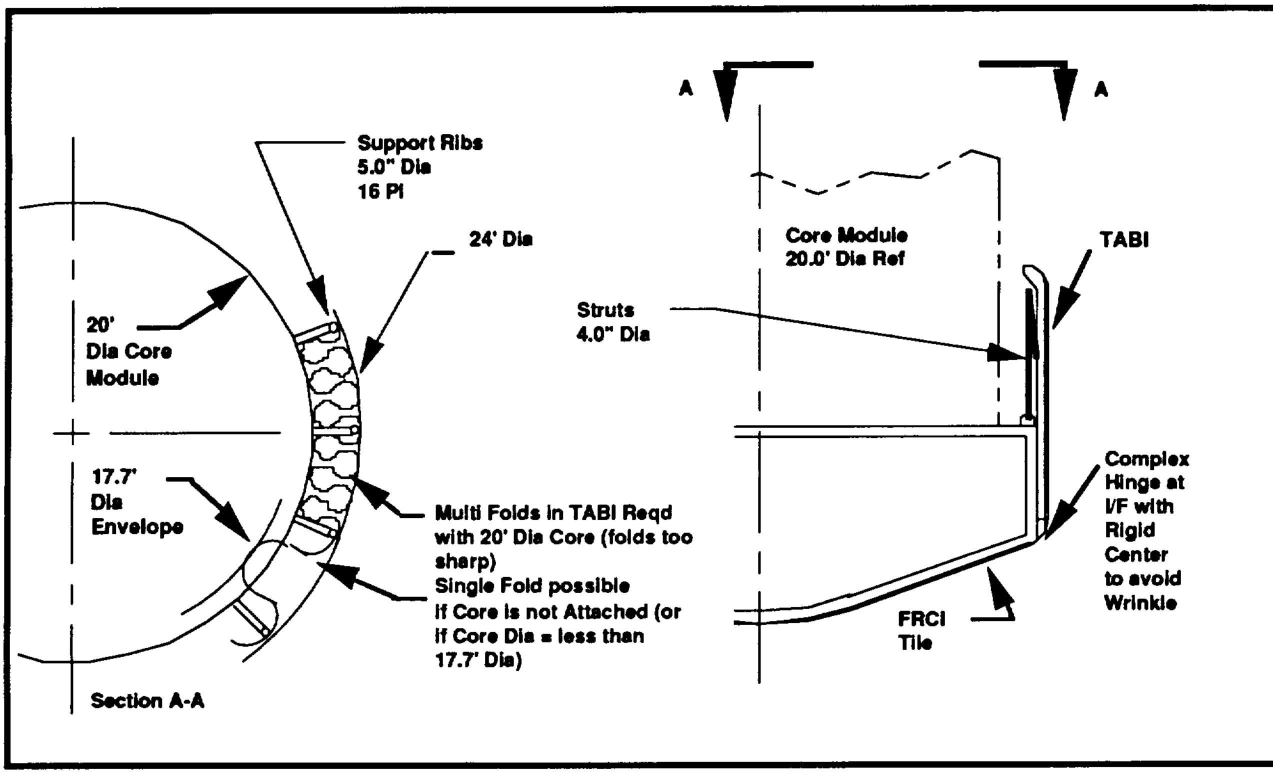


Figure 2.3.4.3.1.1-4 Flexible Aerobrake - Stowed

Figure 2.3.4.3.1.1-5 shows the flexible aerobrake in the deployed configuration. A concern with the flexible concept is how to implement the TABI trailing edge support member. Ideally it would be attached to the edge of the TABI in the stowed (folded) configuration, shown in the previous chart.

This imposes obvious difficulties since it would have to transition from a highly curved shape to a nearly straight one and attain rigidity in the deployed configuration. Inflatable toruses, memory materials, in-place foaming agents, telescoping tubes, and hinged tubes are possibilities, but all have drawbacks. This aspect remains a key issue with this concept.

A key element in the trade between a rigid and flexible system is the deployment of the flexible system once it has been delivered to LEO. The flexible aerobrake is shown in both the stowed

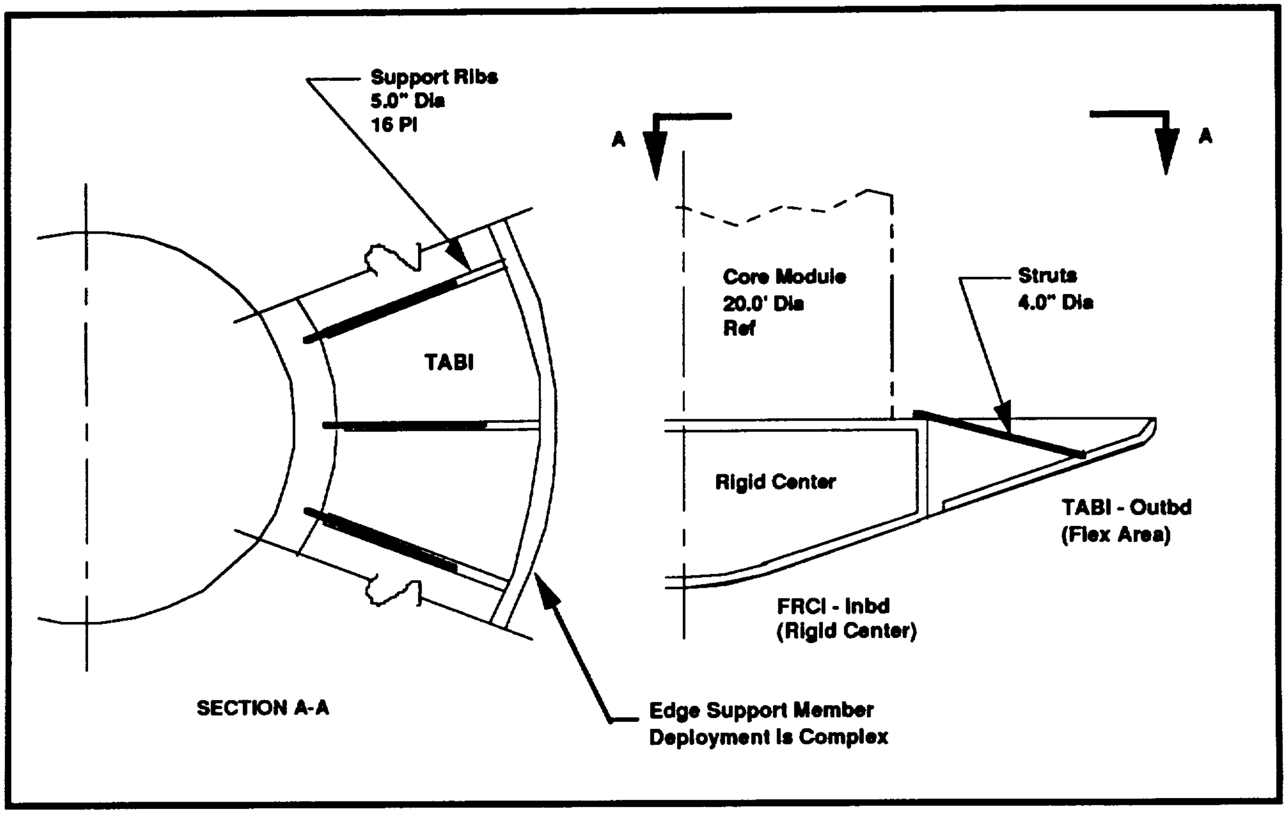


Figure 2.3.4.3.1.1-5 Flexible Aerobrake - Deployed

(folded) and the deployed configurations in Figure 2.3.4.3.1.1-6. The RMS arms engage the rib tips in the stowed position, and after assurance of proper engagement, the robotic center moves forward as the arms extend outward, deploying the flexible portion of the aerobrake. The RMS deployment concept eliminates the need for multiple deployment motors and other devices required for a self deploying aerobrake. This approach therefore minimizes aerobrake weight, but would require that the aerobrake be separate (not attached to the core vehicle at aerobrake deployment) or that an RMS system could unfold the ribs while operating from the forward side of the aerobrake.

A somewhat simpler RMS deployment concept using an umbrella-type mechanism was also considered and found to reduce on-orbit deployment timelines.

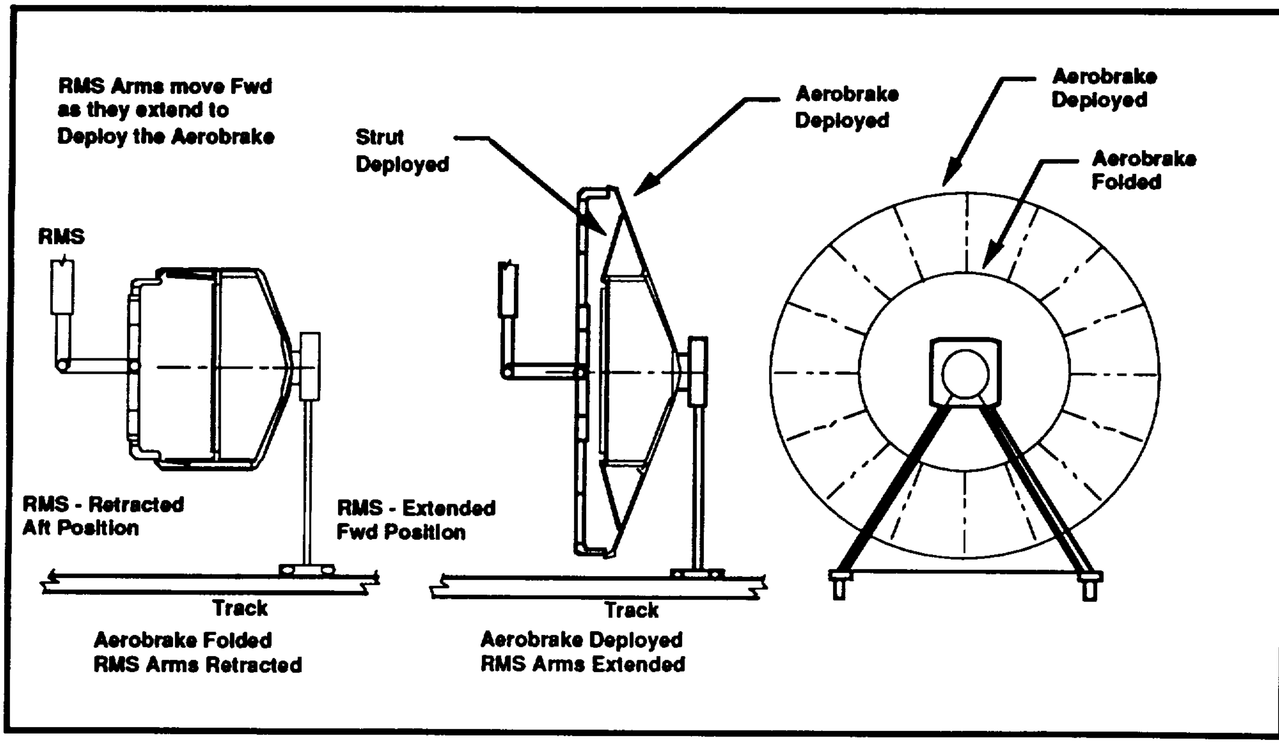


Figure 2.3.4.3.1.1-6 Flexible Aerobrake Deployment

The following paragraphs summarize the key characteristics of the rigid and flexible aerobrake concepts as defined during this study. These comparisons are based primarily on the baseline 8-segment rigid concepts. Implications of the 3-section rigid versions developed late in the study are interjected as appropriate.

Weight - Initial estimates indicated the rigid and flexible concepts had comparable weights. However, the apparent need to increase the TABI thickness for strength purposes and/or the addition of flexible support structure under it would add to the flexible aerobrake's weight. Also, TABI trailing edge support provisions have not been adequately defined. Finally, the higher ballistic coefficient (20BTU/ft^2) now established, the heating rates on the TABI associated with a 45 ft. diameter brake may exceed TABI limits, requiring a larger diameter for a flexible aerobrake than for a rigid one. Thus, the weight comparison now appears to favor the rigid concept.

On-orbit Assembly/Deployment - Handling and installing the panels in either version of the eight panel rigid concept is clearly a more difficult and time consuming task than rotating the 16 hinged ribs into place for the flexible design. Even the hinged 3-piece rigid concept requires more time than does the flexible concept. However, evaluators of on-orbit timelines felt that greater uncertainties existed for the flexible concept operations than for the rigid concepts; see paragraph 2.3.4.3.4.

On-Orbit Inspection/Repair - The continuous nature of the flexible blanket design results in fewer on-orbit-created joints to inspect, but the same feature results in lessened capability for on-orbit replacement of a locally damaged area.

"Reliability" - As used here, this characteristic relates to the possibility that the aerobrake will fail in operation in spite of passing qualification ground and flight tests. "Man Rateability" may be a better term for it. Purely subjective assessments in this category tend to favor the rigid concept primarily on the basis that it should be less sensitive to the unpredictable variations from ground to flight and from flight to flight, e.g.., dynamic loading.

Technology Development Required - Significant technology development is required in both concepts, but there appear to be more areas requiring resolution with the flexible design, e.g., trailing edge support and inspection/repair provisions.

Launch Vehicle Stowage/Space Station Storage - Manifesting the eight-segment rigid concept in the launch vehicle cargo bay or payload fairing requires stacking the eight panels in a separate section of the compartment (the core module is assumed to be attached to the center section of the aerobrake in this configuration). With the flexible concept and the hinged three-piece rigid concept, the aerobrake remains in one piece but must be separated from the core vehicle.

Producibility - More definition of flexible concept fabrication processes is required to adequately compare the two concepts from a manufacturing producibility standpoint. Part count, which would favor the flexible concept, may not be the best yardstick since the complexity of blanket manufacturing and splicing may offset the large number of individual tiles required with the rigid design.

Cost - Comparative life cycle costs have not been estimated.

Evolution - For the larger diameter aerobrakes required for Mars applications the flexible concepts would allow better packaging in the launch vehicle cargo bays. Heating rate limits on the flexible materials, however, would have to be examined for specific cases.

Summary and Recommendations - The rigid aerobrake appears to afford a somewhat lower risk approach based on these preliminary configuration definitions. At this time, however, it appears the potential for simplifying on-orbit assembly, along with the other identified potential advantages, are sufficient to warrant the continued to pursuit of flexible and rigid designs. Also, it appears that further optimization of the hinged rigid three-piece design could result in achieving some of the deployment benefits associated with the flexible concept.

2.3.4.3.2 Guidance and Control—The aerobrake phase of a lunar return vehicle places the spacecraft in a low park orbit around the Earth after transitioning from a lunar return trajectory. The amount of velocity reduction achieved in this maneuver is on the order of 3100 meters/second and must be performed with precise control to prevent reentry or skipout. Previous OTV work conducted in the MSFC sponsored OTV Phase A study established fundamental L/D requirements for lunar return (Orbital Transfer Vehicle Concept Definition and System Analysis Study, Vol XI - Study Extension II Results, MCR-86-2601, NAS8-36108, January 1988). These requirements were arrived at by an assessment of terminal navigation and midcourse correction errors, as well as atmospheric and vehicle aerodynamic flight variations. Study of these errors associated with lunar return established control requirements of 0.14 L/D for insertion into a 454 kilometer circular park orbit. This level of control is assumed to be for trajectory corrections and is not sized to perform significant (>2°) plane change maneuvers. It is felt that large orbital plane misalignments brought about by free return lunar abort scenarios are best accommodated by relatively small secondary propulsion maneuvers in deep space.

The lunar return L/D levels were derived using parametric analyses with margin applied to account for unmodelled rate effects. A series of dispersed test cases are required to gain confidence in the stability of trajectories using this control level. This has been accomplished by utilizing a four degree of freedom (3 translational, 1 rotational in roll) closed loop entry simulation. This simulation is called CLAAS for Closed Loop Aeroassist Simulation and has been used in a number of manned and unmanned aerobraking studies. Critical to the aeroentry process is the impact of atmospheric density shears which can cause excessive dispersions because of unanticipated density shifts at critical phases of entry. The space shuttle has proven to be a good source of atmospheric data because of the recording of accelerometer data in the entry phase. This data has been characterized as density variations from predicted atmospheric conditions. Testing was performed on the lunar configuration with a variety of these STS atmospheric profiles as well as entry targeting and vehicle angle of attack variations. Table 2.3.4.3.2-1 summarizes the results of this testing. The results show good stability in the exit orbit parameters for all the shuttle atmospheres with a worst case circularization velocity requirement of 134 meters/second which is only 12 meters/second higher than the nominal circularization requirement of 122 meters/second for a 454 kilometer orbit. A representative atmosphere (STS-4) was chosen for further testing in conjunction with angle of attack and targeting errors as shown in the bottom of the chart. Here again the stability is good with the worst case circularization Δ-velocity of 146 meters/second (24 meters/second above nominal). These results indicate good stability for the 0.14 L/D configuration.

Table 2.3.4.3.2-1 Lunar Return with Load Relief

LUNAR RETURN WITH LOAD RELIEF, L/D=0.14, W/CdA=10.8 LB/FT2

	DISPERSION	EXIT ORBIT PERIGEE (NM)	EXIT ORBIT APOGEE (NM)	ΔV TO REACH PARK ORBIT (FPS)*	PEAK LOADS (g's)	PEAK HEATING** (BTU/FT2-SEC)	INTEGRATED HEATING** (BTU/FT2)
942	NOMINAL	19.3	244.8	398.9	3.52	200.4	22843
943	STS-1	15.4	244.6	406.5	3.64	202.5	22529
944	STS-2	18.1	231.1	424.8	3.24	199.9	23886
945	STS-3	19.4	250.4	407.3	3.41	197.5	22746
946	STS-4	16.8	242.9	406.7	3.04	192.8	23618
947	STS-5	-2.8	244.8	439.7	3.83	207.3	21702
948	STS-6	24.6	264.1	420.7	3.73	207.3	22506
949	STS-7	18.1	255.6	418.5	3.43	187.7	23028
950	STS-8	22.0	245.4	394.3	3.46	203.8	23004
951	STS-9	25.2	265.8	422.5	4.09	207.2	22608
952	STS-11	24.1	255.3	406.9	3.31	189.6	23366
953	STS-13	16.8	250.9	413.0	3.73	201.3	22433
954	STS-14	13.1	252.5	422.5	3.63	202.4	22097
957	ΔPER = +0.28 nm + STS-2	2.9	216.3	478.7	4.03	199.3	22465
958	ΔPER = -0.28 nm + STS-2	18.3	224.3	436.3	3.58	197.4	23536
955	Δ ALPHA = +2.0° + STS-2	18.3	233.8	419.8	3.47	198.8	23965
956	Δ ALPHA = -2.0° + STS-2	19.8	250.5	406.9	3.83	198.2	22524

* FINAL PARK ORBIT OF 245 X 245 NM IS REACHED VIA ΔV1 AT APOAPSIS FOLLOWED BY ΔV2 AT PERIAPSIS (ΔV=ΔV1+ΔV2)

** CONVECTIVE HEATING RATES FOR 1 FT SPHERE

An additional issue for the entry of an aerobraked vehicle is the attitude stability when performing roll maneuvers. Although many previous flight vehicles have utilized control their entries, the control stability of any particular vehicle is configuration dependent. Although a full six degree of freedom control analysis is required to definitively validate complete control behavior, it is possible with simpler analysis to assess basic stability. An analysis was performed which looked at the roll jet interaction with the aerodynamic properties of the 45 foot blunt aerobrake. Because the vehicle enters the atmosphere at an angle of attack, the symmetry axes do not correspond to the aerodynamic axes. For the LTV lunar return vehicle there exists a 9.34° angle between the two coordinate systems. This offset results in coupling between the roll actuation jets and a resulting torque into vehicle yaw. As an example, the use of twin 100 pound roll thrusters yields a torque of 2664 foot-pounds in the wind relative roll axis and a coupled 438 foot-pounds in the yaw axis. Computation of aerodynamic restoring torques in the yaw axis shows that for a 45 ft aerobrake at peak load condition, a sideslip angle of only 0.05° is sufficient to produce an equivalent torque. Thus a total oscillation in the yaw axis of ±0.10° would be expected due to the misalignment of the roll thrusters. Although lower dynamic pressures would result in larger magnitude oscillations, these only become significant at the very beginning of the entry. For this phase of flight the vehicle is in a 3-axis attitude hold anyway because of low aerodynamic pressures. This indicates that the use of body-aligned thrusters in an angle of attack aeroentry configuration is not a major concern for the lunar return LTV. Although not incorporated in this analysis, the influence of aerodynamic damping terms, and the normal operation of the yaw rate feedback control would act to null out this disturbance as well.

2.3.4.3.3 Configuration and TPS Material Response Sensitivities - Based on previous aerobrake configuration trade studies, guidance and control corridor requirements studies and aerodynamic and aeroheating analyses, the 45 foot diameter, 70° half angle symmetric blunted cone configuration with an L/D of 0.15 at 10° angle of attack was selected as the baseline for the LTV mission studies. This shape, based on original core vehicle mass property and sizing estimates, resulted in a ballistic coefficient or W/C_DA, of 14 pounds/square foot. Also, vehicle dimensions were compatible with the predicted wake closure or impingement angle of 22°, see Figure. 2.3.4.3.1.1-2, which was based on experimental ground test data for a similar geometry aeroshell. Refinement of core vehicle mass properties, however, has caused the W/C_DA to increase to approximately 20 pounds/square foot· thus increasing peak heating. This, together with some preliminary aerophysics CFD results from LaRC indicating that wake impingement angles may be larger than the 22° established earlier, suggest the reassessment of the aerobrake diameter. The nose radius influence on heating was included in this assessment.

Analysis - Figure 2.3.4.3.3.2-1 illustrates the sensitivity of the total heat flux and the convective and radiative heat flux components to the radius of curvature of the spherically blunted 70° half angle cone aerobrake. The total heat flux is minimized at a radius of 22.5 feet at a base radius (R_B) to nose-radius ratio (R_N) of 1.0. With this geometry the radiative heat flux is 1/3 of the total compared to a much smaller fraction of the total for the baseline R_B/R_U ratio of 2.0. This has TPS material response implications but the larger nose radius helps to alleviate concerns with RPS material heat flux/temperature limitations.

Next, the W/C_DA has an influence on surface temperature through its influence on heat flux. This is illustrated in Figure 2.3.4.3.3.1-2. The range of temperatures shown for a given W/C_DA value exists because of unknowns in some of the basic TPS material characteristics as identified on the plot. The larger radiative component mentioned earlier plays a role in the size of this uncertainty band since absorbtance properties are wave length dependent and not well characterized. However, W/C_DA values of the order of 15 pounds/square foot are seen to produce significantly lower surface temperatures. The horizontal bars represent estimates of current material limits for single and multiple uses of the aerobrake.

The next three charts, Figures 2.3.4.3.3.1-3 thru 5, illustrate some of the implications of producing lower W/C_DA values by increasing aerobrake diameter. In the first of these, Figure 2.3.4.3.3.1-3, it can be seen that the launch vehicle payload bay diameter must also increase to accommodate the larger aerobrake. The values shown are for the hinged 3-piece rigid aerobrake and would not increase with aerobrake diameter as rapidly for the other configurations.

In the next chart, Figure 2.3.4.3.3.1-4, the benefits of a larger diameter aerobrake are seen in the growth of the angle between the tip of the aerobrake and the aftmost surfaces of the core vehicle, i.e., with a 53 foot diameter aerobrake a wake impingement angle of 32° could be tolerated. This is of interest because preliminary CFD analyses by LaRC suggest the impingement may be as great as 28° (for a 10° angle of attack).

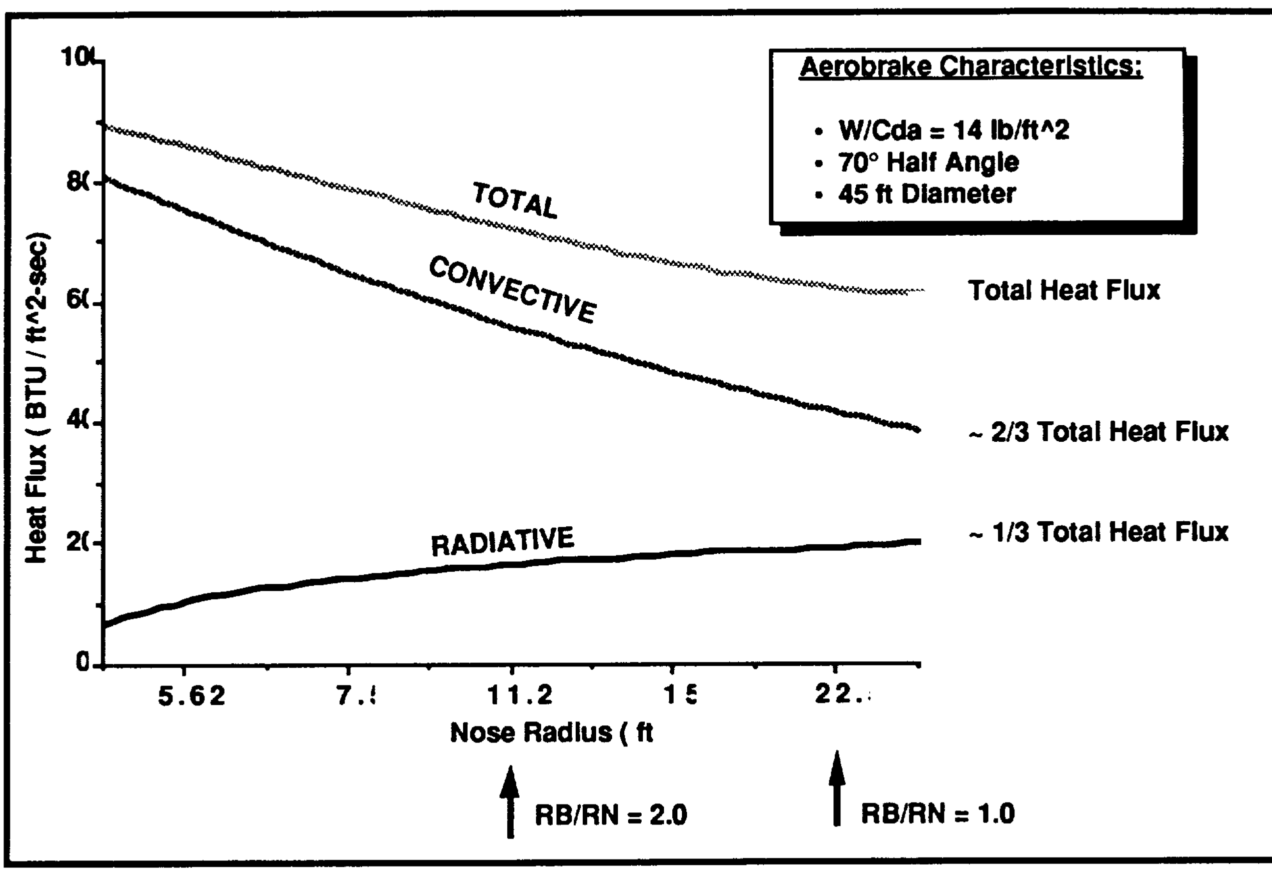


Figure 2.3.4.3.3.1-1 Heating Component Sensitivity to Nose Radius

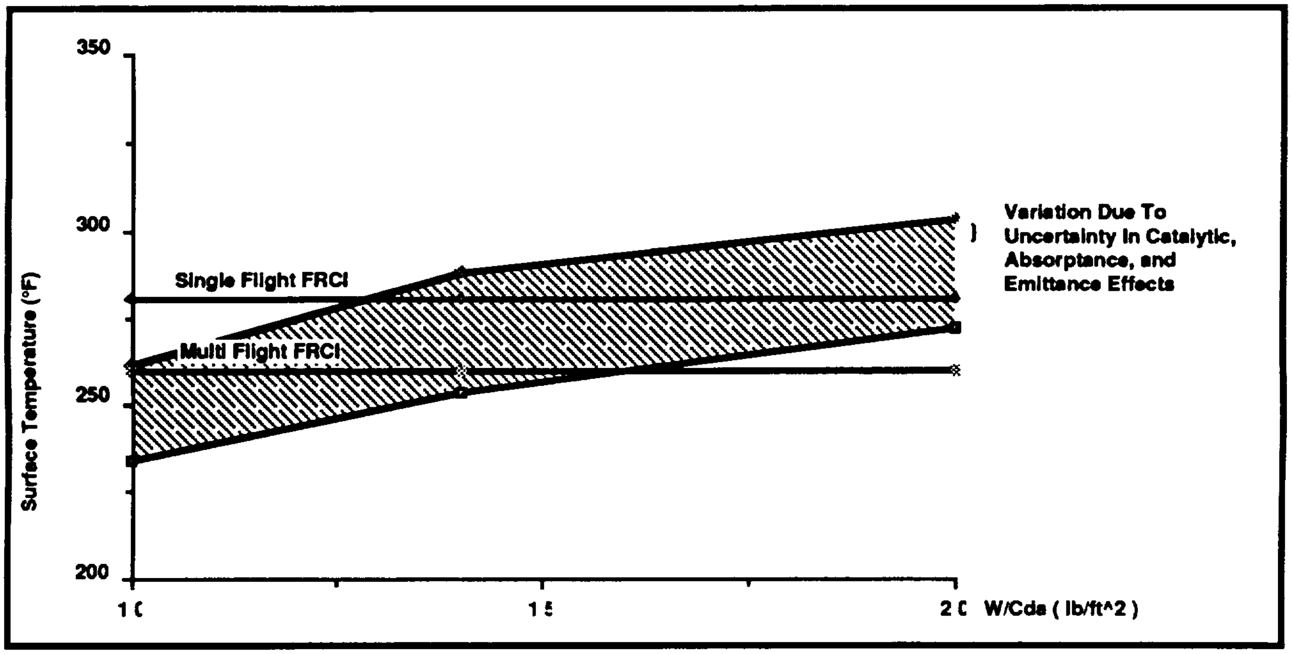


Figure 2.3.4.3.3.1-2 Surface Temperature Sensitivities (RB/RN = 1.0)

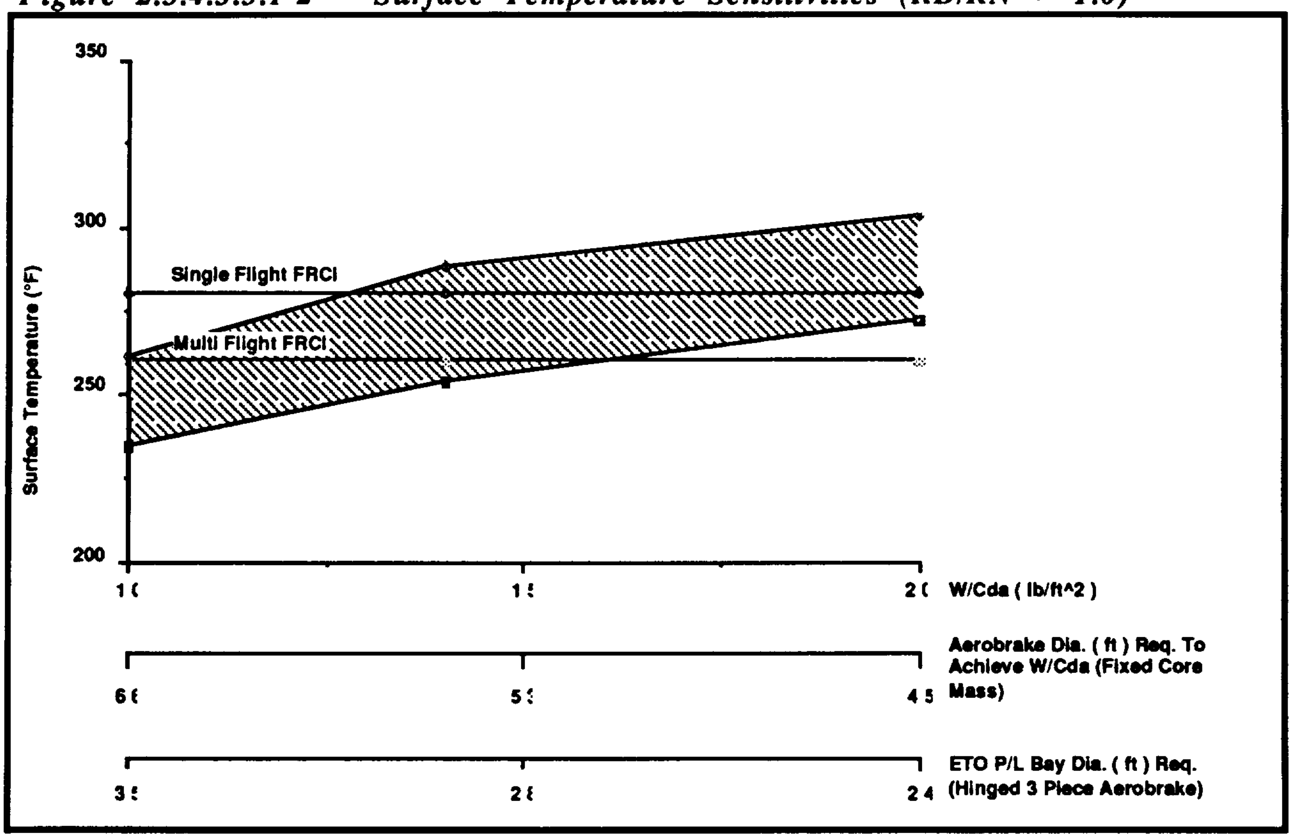


Figure 2.3.4.3.3.1-3 Surface Temperature Sensitivities (RB/RN = 1.0)

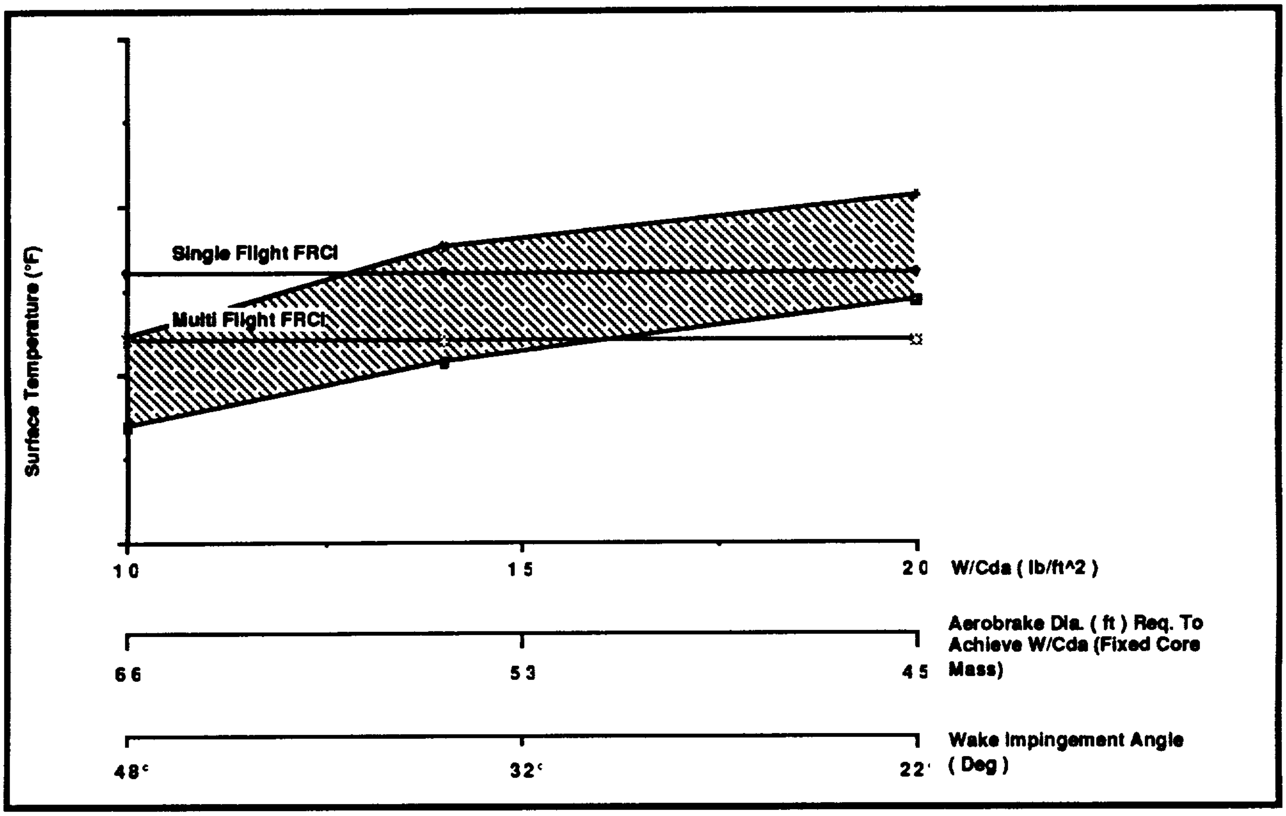


Figure 2.3.4.3.3.1-4 Surface Temperature Sensitivities (RB/RN = 1.0)

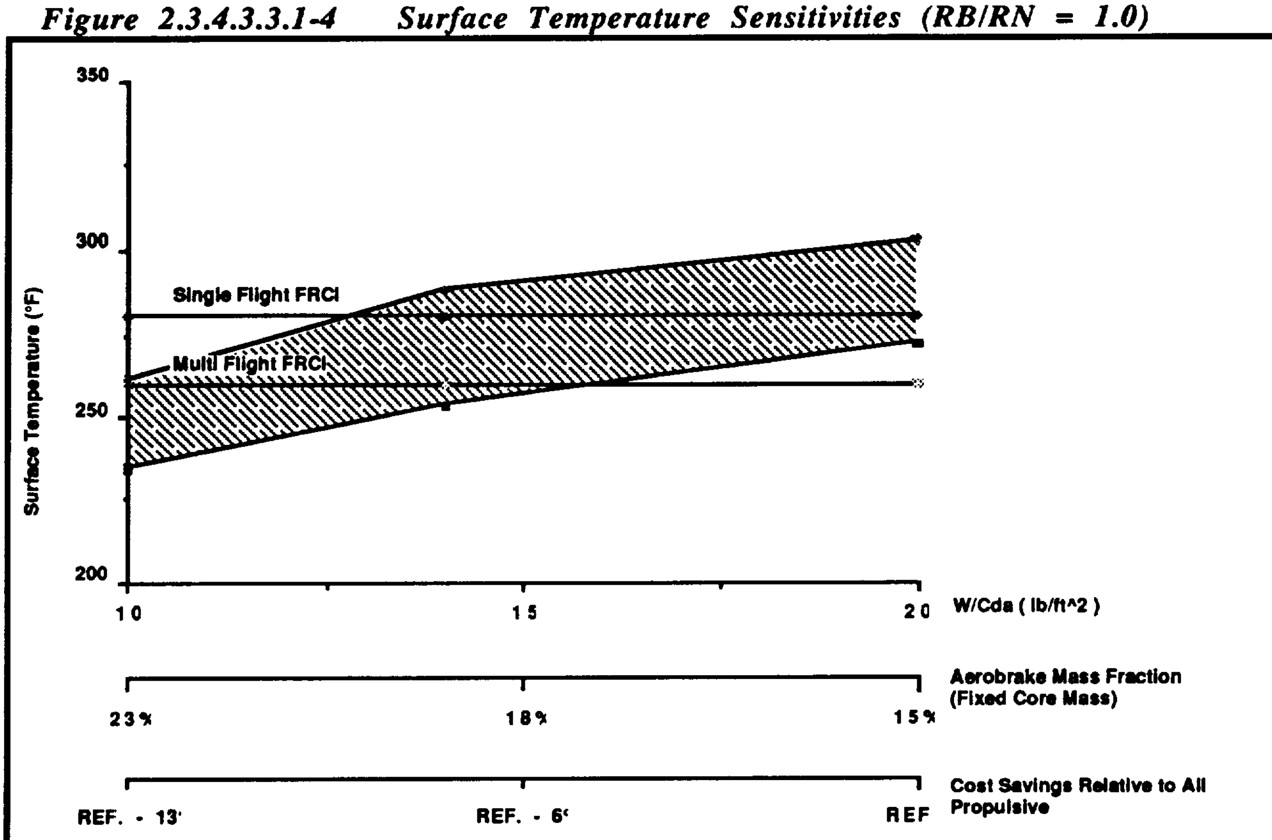


Figure 2.3.4.3.3.1-5 Surface Temperature Sensitivities (RB/RN = 1.0)

Lastly, Figure 2.3.4.3.3.1-5 attempts to assess the cost penalty associated with greater aerobrake mass fraction produced by the larger diameter. The reduction in cost savings relative to an all propulsive system is derived from the trade study data of paragraph 2.3.3.1 and reflects the findings from that analysis that for moderate departures from the baseline aerobrake mass fraction cost benefits relative to an all propulsive system are not significantly degraded.

Summary and Recommendations - To minimize front side heating levels and provide greater shadowing effect for the core vehicle a slightly larger, 50 foot to 54 foot diameter aerobrake would appear to be desirable. A larger nose radius, $R_B/R_N = 1.0$, would also be desirable pending evaluation of implications on aerodynamics and increased radiative heating component contribution.

2.3.4.3.4 On-Orbit Operation Comparison Analysis—The objective of this study was to compare the on orbit operations timelines for various rigid and flexible/rigid aerobrake configurations. Considered in this analysis were seven different aerobrake configurations that included four rigid and two flexible Martin Marietta Corporation (MMC) designs and one rigid MSFC design. The designs consisted of:

- Rigid eight panel segments with integral ribs (R-1)
- Rigid eight panel segments with folding ribs hinged to the center section (R-1A)
- MSFC rigid eight panel segments similar to MMC R-1 configuration (MSFC Rigid)
- Rigid separate three piece aerobrake fastened along common bulkheads (R-2)
- Rigid hinged three piece aerobrake fastened along common bulkheads (R-3)
- Flexible/Rigid Core aerobrake deployed by multiple telescoping arms (F-1)
- Flexible/Rigid core Aerobrake with umbrella type deployment (F-1A)

Analysis - This study, performed by the McDonnell Douglas Company under subcontract to the Martin Marietta Corporation, is based on MMC configurations. A more detailed account of the results is provided in the Volume II Appendix. The groundrules used in developing these timelines are consistent for all seven configurations evaluated. These groundrules/assumptions differ somewhat from those used in an earlier aerobrake assembly study which examined the question of IVA/telerobotic assembly versus. EVA assembly based on the 90 Day Study rigid aerobrake configuration. Therefore comparisons of assembly times for these seven configurations with those of the earlier study are not appropriate. The key ground rule in developing timelines for assembly of the seven designs is that IVA/telerobotics will be used with EVA as backup only. This choice was somewhat arbitrary and subject to further assessment. It is believed to provide a valid comparison of the assembly timelines. An interesting aspect of this study is the assessment of what portion of the assembly time cannot be accomplished while the protecting enclosure (ASF) is in place around the aerobrake operations area, i.e., how much of the assembly time is "unprotected." Other groundrules and assumptions are listed in the Appendix.

An example of the specific elapsed times for each step of the assembly and inspection process for the three-piece hinged aerobrake can be found in Figure 2.3.4.3.4.1-1. Similar timelines were developed for each of the other configurations and the results are summarized in Figure 2.3.4.3.4.1-2. The flexible configurations are seen to require less assembly time due to their not requiring panel alignment and attachment operations nor TPS closeout operations. The three-piece rigid aerobrake concepts, however, are assumed to require TPS closeouts, i.e.., installation of separate, mechanically attached TPS panels over joints between the three structural sections and thus take longer times. It may be feasible to preclude these TPS closeout panels, particularly with the hinged version of this concept. The eight-panel rigid designs are the most time consuming due to the greater number of joints and the lack of the self contained alignment features of the hinged three-piece rigid design.

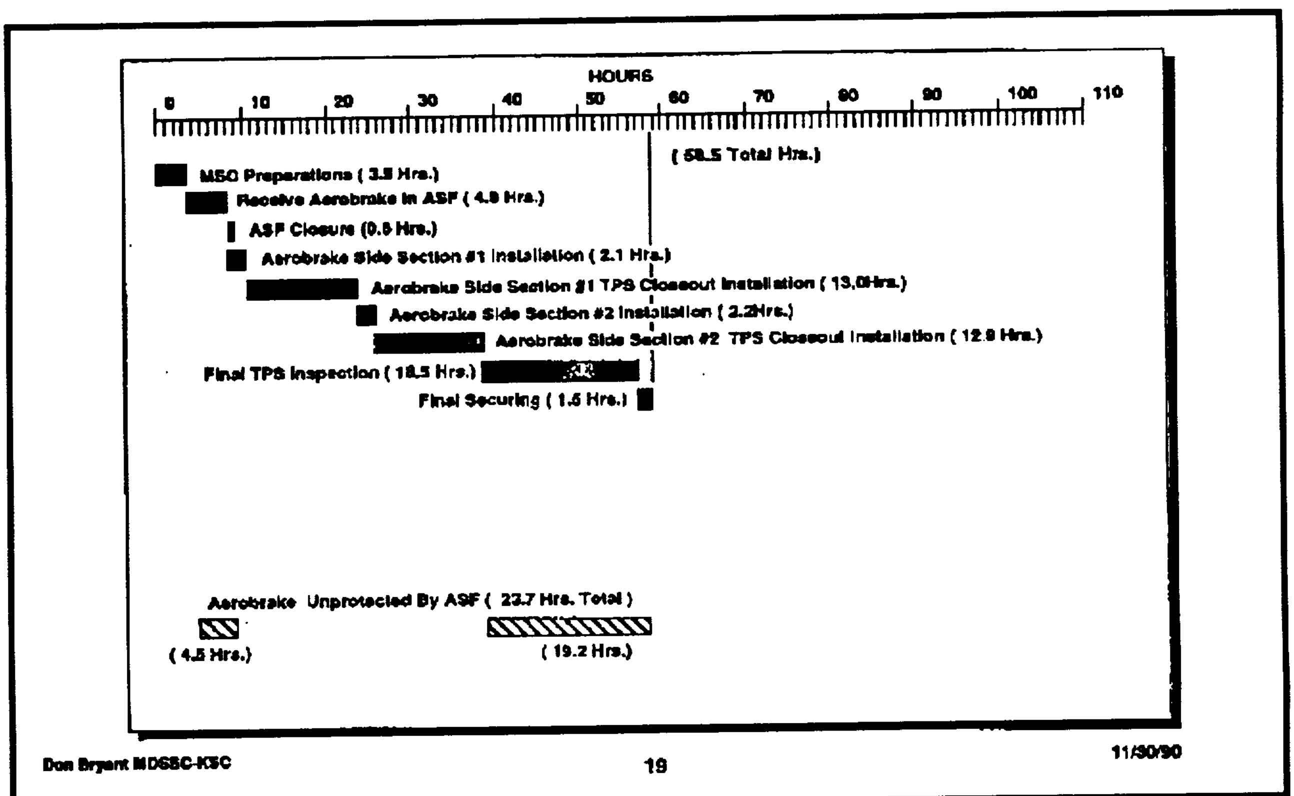


Figure 2.3.4.3.4.1-1 Assembly Timelines for the Rigid Three-Piece Hinged Aerobrake, MMC-R-3

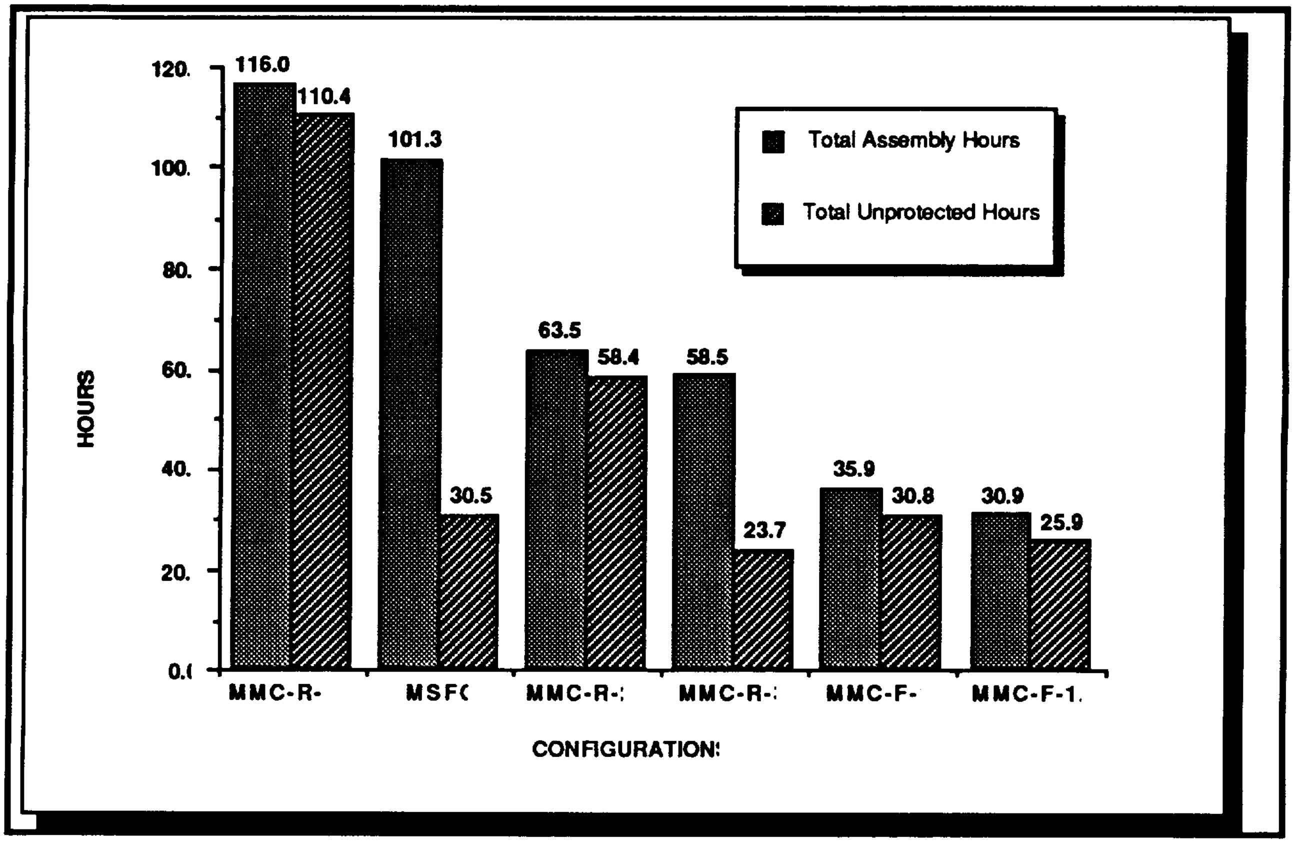


Figure 2.3.4.3.4.1-2 Aerobrake Assembly Timeline Comparison

The differences in unprotected hours shown in Figure. 2.3.4.3.4.1-2 reflects the assessment of how many and what size RMS devices must be engaged in bringing segments of the aerobrake together and holding them in place while implementing the joining functions. The hinged three-piece design is estimated to have a slight advantage in this regard.

It should be pointed out that the analysts placed a greater uncertainty on the estimates for the flexible configurations. This was due at least in part to less complete definition of the flexible aerobrake design and its deployment process. It should also be noted that NASA Langley personnel involved in space crane development design and testing caution against anticipating that assembling a large number of smaller segments (as with the eight-segment concept) will necessarily be a more difficult and time consuming task than assembling a few large segments. Their concerns are with alignment of long joints and the implications of emergency stops and/or the significant damping times associated with joining large masses.

Summary and Recommendations - From the standpoint of the initial deployment/ assembly operations, the flexible aerobrake concept as envisioned in preliminary configuration definitions will take less time to accomplish. The rigid three-piece hinged concept is next in order, and, with the incorporation of self sealing, TPS joints could become a close competitor.

Assembly operations for both concepts should continue to be studied with emphasis in the rigid concept on self latching structural joints and self sealing TPS joints. Repair/replacement provisions need to be established for both types of construction.

2.3.4.4 Structures Analysis—The structures analysis and study activity conducted in the STV Study program provided an in-depth assessment of the LTS structural material and design configuration. The primary area of focus surrounds the design and material selection for the propellant tanks. These areas represent a significant impact on the overall transportation system weight, manufacturing, and LEO assembly requirements.

2.3.4.4.1 Intertank Configuration versus Nested Domes Trade Study—The study objective was to compare a nested dome tank configuration with the baselined intertank configuration to determine which was most effective and efficient. Key issues addressed were weight, cost, and producibility. The basic system impact is the performance of the vehicle as affected by the weight of the various components, one of which is the propellant tanks. A nested dome configuration requires less space in the cargo bay than the baselined intertank configuration thus contributing to a vehicle weight reduction.

The groundrules and assumptions defined for this study included physical and production characteristics.

- Baseline Configuration - Tankset with Isogrid Construction, 0.707 inch elliptical domes
- Affected Area - LO_2 & LH_2 domes and connecting structure
- Material - Weldalite™ for both configurations, graphite-polyimide honeycomb used for the intertank
- Nested domes - spherical dome geometry

2.3.4.4.1.1 Analysis—The methodology used to analyze both configurations consisted of two phases. The initial phase produced a recommended design for both the tank domes and the interconnecting structure for the intertank and the nested dome configurations. The recommended intertank design is shown in Figure 2.3.4.4.1.1-1, the design for the nested dome configuration is shown in Figure 2.3.4.4.1.1-2. In the second phase, these designs were evaluated for weight, estimated cost impacts, schedule impacts and constraints, and tooling impacts.

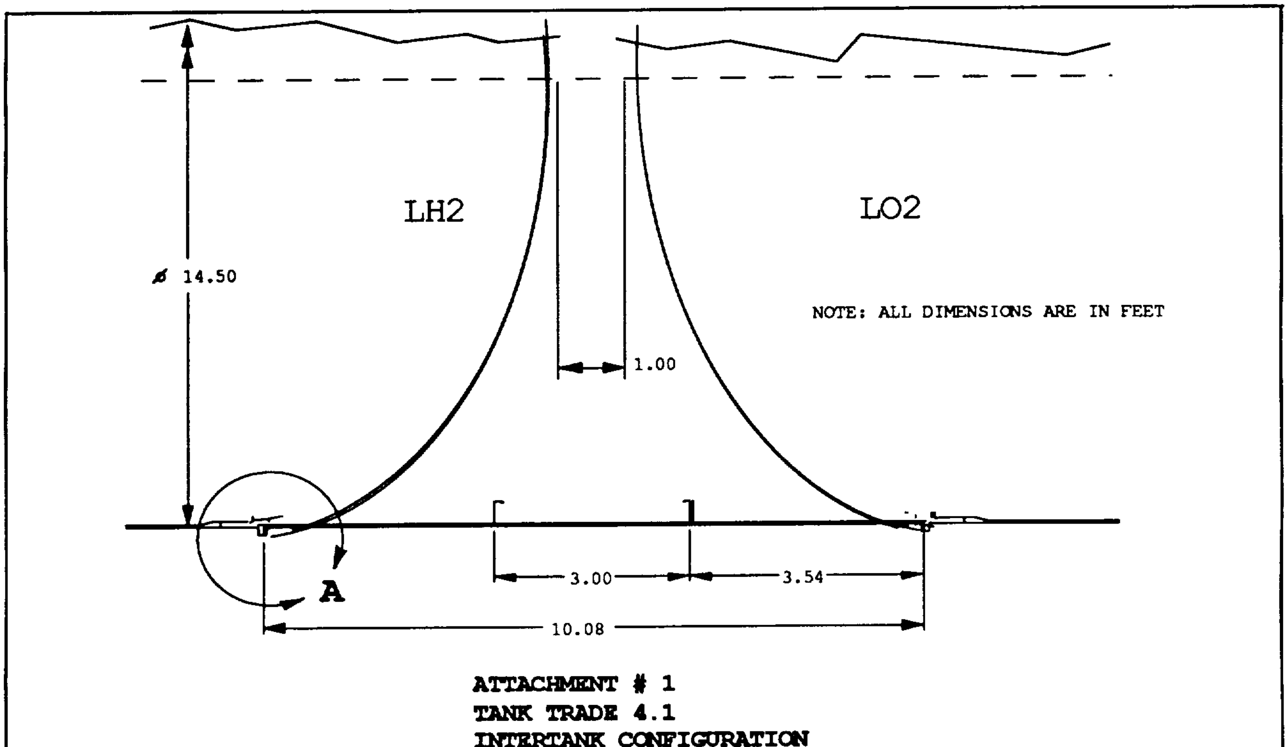


Figure 2.3.4.4.1.1-1 Intertank Configuration

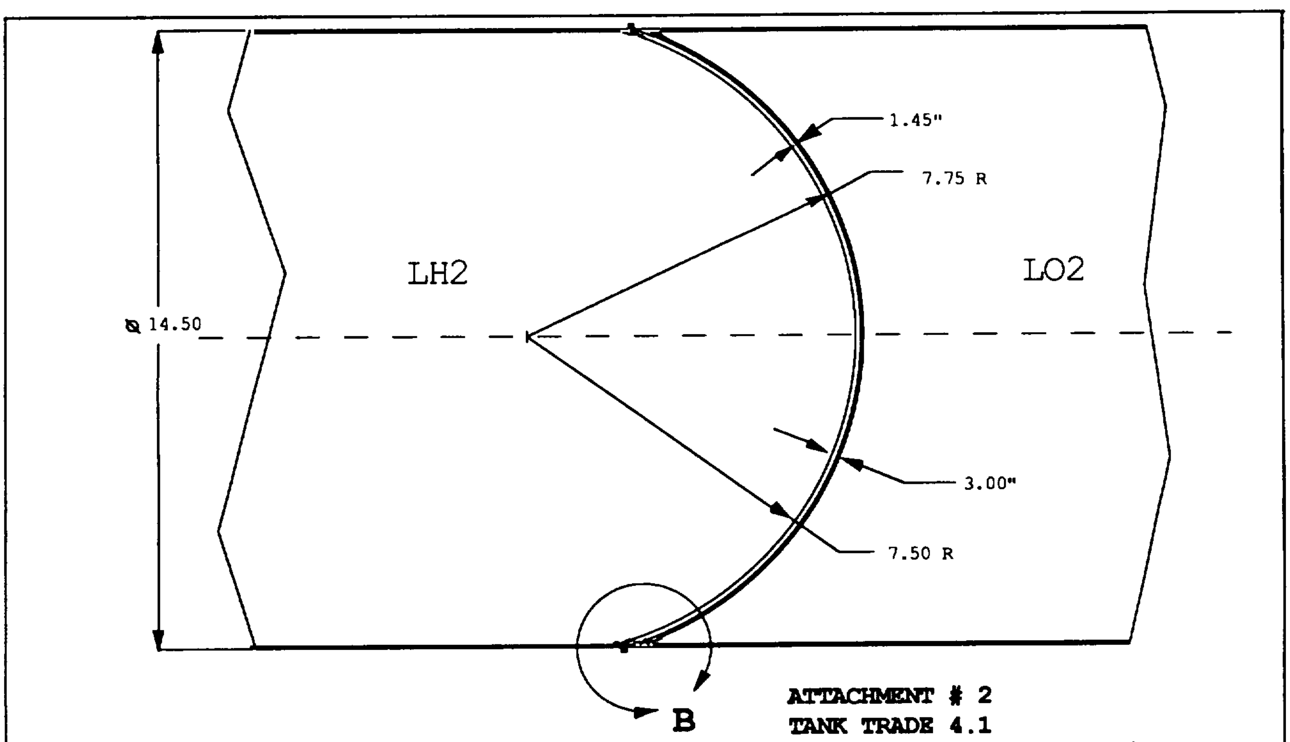


Figure 2.3.4.4.1.1-2 Nested Dome Configuration

Summary and Recommendations - The study's recommendation is that the intertank configuration remain the baseline design since the small weight reduction provided by the nested dome configuration does not offset the additional schedule risk and manufacturing difficulties anticipated with the nested dome configuration. Table 2.3.4.4.1.2-1 summarizes the results of this evaluation.

Table 2.3.4.4.1.2-1 Summary of Evaluation Results

	BASELINE Intertank (I/T) Configuration	Alternate Nested Dome Configuration	Comments
Weight Impac	Honeycomb Intertank Simple Dome Const Honeycomb Barrel Simple TPS A/R	No I/T - Short Conn Sect Beef-up Concave Dome Beef-up adjacent Barrel Honeycomb Insulation	Nested Domes will provide a small weight reduction
Cost Impac	Honeycomb Intertank Common Spun Domes Honeycomb Barrel Simple TPS A/R	Two Spherical Spun Dome configurations Isogrid LOX Barrel Honeycomb Dome Insulation	Special Insulation reqmts for domes but no I/T with the Nested Dome Design
Schedule Impac	Medium Risl	Medium to Moderate Risk	Some additional Schedule Risk with Nested Config
Manufacturing Impac	No Serious Impacts Anticipated	Beef-up and Insulation Application & Inspection Problems	Fewer Manufacturing problems anticipated with the Baseline Configuration
Tooling Impac	Intertank Tool reqd All Domes built on Common Tools No Serious Impacts Anticipated	Additional Dome Tools Required for spinning and machining domes	Nested Dome Tooling will be more complex but No I/T Tooling reqd fo Nested config

2.3.4.4.2 Weldalite™ versus 2219 Aluminum Alloy Material Trade Study—The study objective was a comparison of 2219 Al Alloy with the baselined Weldalite™ to determine the most cost effective structure. Key issues addressed were weight, cost, and producibility. The basic system impact is manufacturing the various vehicle components, one of which is the propellant tanks. Due to the near term cost of Weldalite™, a trade on the weight benefits of Weldalite™ against a more cost effective method of manufacturing propellant tanks was suggested.

The groundrules and assumptions defined for this study were:

- Baseline material - Weldalite™

- Alternative material - 2219 Aluminum Alloy
- Affected items - Propellant tank elements/pressure vessel elements (spun domes, dome chords, isogrid barrels)
- Minimum isogrid pocket skin thickness = 0.050 inches
- Minimum isogrid rib thickness = 0.063 inches (limited by manufacturing constraints)

Analysis - The analysis was conducted in two phases. The initial phase produced a recommended tank set design using both Weldalite™ and 2219 aluminum alloy material. In the second phase the designs were evaluated for weight, estimated cost impacts, schedule impacts and constraints, and tooling impacts.

Table 2.3.4.4.2.2-1 Summary of Evaluation Results

	BASELINE Weldalite Alloy	ALTERNATE 2219 Alloy	Comments
Weight Impact	Reference	2219 Tank Set = 6 -10 % weight increase. Individual elements = 6 % to 15 % increase	An All-2219 Tank Set will be approx 10 % heavier
Cost Impact	High Material Cost	Less Expensive Material	Weldalite material costs will be 3 to 4 times higher per lb, but less mat'l req'. than for 2219
Schedule Impact	Medium Risk	Low Risk Mat'l readily available	Less Risk with 2219 due to mat'l availability
Manufacturing Impact	Welding techniques now being developed. No problems anticipated	Welding techniques are already well defined	No significant mfg impact anticipated
Tooling Impact	Will require some new weld tooling	No new tooling problems anticipated	2219 MAY impose less impact to tooling

Summary and Recommendations - The recommendation emerging from this study is that further analysis will be required as the configuration definition matures. If weight/performance is most critical, Weldalite™ should be incorporated into the design since it represents a weight saving potential over 2219 aluminum alloy as well as processing increased mechanical properties.

If material cost is key, 2219 aluminum alloy should be incorporated into the design because of its manufacturing cost advantages, which have been established through proven manufacturing techniques and tooling requirements. An alternate approach would be to use Weldalite™ for the more highly stressed components and 2219 aluminum alloy where section properties are believed to be more important than mechanical properties. Table 2.3.4.4.2.2-1 summarizes the results of this evaluation.

2.3.4.5 Crew Module Analysis—The analysis and study activity performed against the crew module, provided the operational and design data incorporated into the final LTS configuration recommendations. The primary areas of focus involved the basic configuration of the crew module itself as well as specific operational concerns addressing crew visibility. Results of these studies include LTS crew module configurations as well as key life support and safety issues relative to operation and rescue.

2.3.4.5.1 Crew Module Configuration Trade Study—The study objective was to select an overall configuration for the crew module(s) best suited for the LTS mission. The key issues addressed focused on whether the crew module(s) require a new design, a modification of the Apollo design; one or two modules; or a hybrid version being developed as part of the LTS; and whether the LTS crew module(s) should incorporate an EVA air lock or if depressurizing the entire cab would be necessary. In addressing these issues, an assessment of the operational scenarios determining crew module quantities based on nodal operations - such as rendezvous and docking functions in Low Lunar Orbit (LLO) - and determining the sensitivities of differing crew module configuration to mission scenarios, the operational concepts, and demonstrated growth capabilities were considered.

The groundrules and assumptions defined for this study included physical and production characteristics.

a) New versus Modified Apollo Crew Module
   - Rockwell Command Module (CM)
   - Grumman Lunar Excursion Module (LEM)
b) One versus Two versus Hybrid Module
   - Separate Module as defined in the NASA 90-Day Report
   - Common Module is the MSFC LTV baseline shown in the 90-Day Report, modified to support both rendezvous and docking and lunar landing
   - Hybrid module is the Boeing configuration presented at the December 14, 1989 Interim Review at MSFC.

c) Blow Down versus EVA Air Lock
   - Total volume of the crew module will be depressurized if airlock is not used.

Analysis - The analysis methodology approach to analyze the crew module configurations was comprised of three primary phases. Phase I addressed the feasibility of developing a new module versus using the Apollo design, Phase II optimized the quantity of modules, one, two, or a hybrid configuration; and Phase III defined the module sensitivities of mass and volume based on depressurization or addition of an EVA airlock.

New versus Modified Apollo Crew Module - Determining the feasibility of a new module versus the Apollo design required the assessment of both configurations based on mission applicability, man rateability, and a qualitative cost comparison. The results of this assessment are shown in Table 2.3.4.5.1.1.1-1.

Table 2.3.4.5.1.1.1-1: New vs Modified Apollo Crew Module Trade Results

	LTV		LEV	
	New Module	CM Derivative	New Module	LEM Derivative
Applicability to Current Lunar Scenario	Excellent	Poor	Excellent	Good
Applicability to Mars Growth	Good	Poor	Good	Poor
Use as Crew Rescue Vehicle for Direct Return From Moon	Poor	Excellent	Poor	Poor
Man Rating	Man Rating Process Initiated With Earliest Design Effort	Must be Reworked to Incorporate New Man Rating Rqmnts	Man Rating Pro- cess Initiated With Earliest Design Effort	Rework Rq'd to Incorporate New Man Rating Rqmnts
Cost Comparison	—	Some Minor Cost Savings Anticipated	—	Minor Cost Saving Anticipated

The CM Was Designed as a Reentry Vehicle, Not Just a Crew Module. The LEM Crew Module Was Integral to its Vehicle Structure. Trade Should More Appropriately be Made Between LEV vs. LEM.

One versus Two versus Hybrid Crew Module - The evaluation of the quantity and type of crew module required to support the lunar missions, focused on the mass of the module both at LLO and the lunar surface as well as operational differences at LEO, LLO, and the lunar surface. As part of this analysis the interfaces which were considered were: effects between the module; between the module and core vehicle; the transfer of crew impacts; command and control requirements; and the impact of transitioning the vehicle to a cargo only configuration, see in Tables 2.3.4.5.1.1.1-2, -3, -4.

Table 2.3.4.5.1.1.1-2: One vs. Two vs. Hybrid Crew Module Trade Results

Parameter	Separate Modules as Shown in Baseline	Common Module for Both LTV/LEV	Hybrid Module, As Presented by Boeing on 12/14/89 IPR
I/F With Core Vehicle	Hardline Connections for All Support Functions. Can Be Used As a Structural Support Member.	Must Provide Quick Disconnects for All Support Functions.	Must Provide Quick Disconnects for Any Functions Required Between Module Sections.
Crew Transfer	IVA	None	IVA
Crew Module Overall Mass to LLO	10,175kg	6,587kg	6,958kg
Crew Module Mass to Lunar Surface	4,388kg	6,587kg	4,388kg
Minimum Number of Command & Control Stations Required	2	1	1
Cargo Only Flight Effects	Both Modules Replaced by One Cargo Module.Transfer System Rq'd.	Common Module Replaced By Common Cargo Module. Transfer System Might Require Adaptation.	Both Portions Replaced By One Cargo Module. Transfer System Required.

Table 2.3.4.5.1.1.1-3: One vs Two vs Hybrid Crew Module Trade Results (cont.)

	Separate Mod- ules as Shown in Baseline	Common Module for Both LTV/LEV	Hybrid Module, as Presented by Boeing at 12/14/89 Review
LTV Crew Module (W/O Crew)	5,787kg	5,787kg	2,570kg
LEV Crew Module (W/O Crew)	3,588kg	-	3,588kg
Crew + EVA Equipment	800kg	800kg	800kg
Crew Module Overall Mass to LLO	10,175kg	6,587kg	6,958kg
Crew Module Mass to Lunar Surface	4,388kg	6,587kg	4,388kg
Delta Propellant Required (LEO TO LLO)	53,927kg	34,911kg	36,877kg
Delta Propellant Required (LLO TO LS)	4,388kg	6,587kg	4,388kg
Total Delta Propellant Required for Crew Module	58,315kg	41,498kg*	41,265kg*

*The Hybrid System Does Not Provide a Significant Improvement Over the Common Module.
(Mass traceability notes shown on next page)

Table 2.3.4.5.1.1.1-2: One versus Two versus Hybrid Crew Module Trade Results (concluded)

Choice of Module Based on Vehicle Configuration

	LTS Configuration	Separate Modules	Common Module	Hybrid Module
Crew Only Configuration	Transfer Vehicle With Separate Landing Vehicle	Best		
	Combined Transfer and Landing Vehicle		Best	
Cargo Only Configuration	Transfer Vehicle With Separate Landing Vehicle		Best	
	Combined Transfer and Landing Vehicle		Best	
Crew and Cargo Combination	Transfer Vehicle With Separate Landing Vehicle		Best	
	Combined Transfer and Landing Vehicle		Best	

EVA Airlock versus Depressurizing Cabin - This analysis was needed to determine the mass and volume sensitivity of the crew module equipped with a system that uses an EVA air lock or one that depressurizes the cabin during EVA activities. The methodology used in performing this study developed a point of departure baseline comparing the volumes of the different crew modules with the volumes of the SSF airlock system (Table 2.3.4.5.1.1.3-1). This database allowed determination of the number of repressurizations required for planned and contingency EVA, the total LO_2/LN_2 volumes required for repressurizing airlocks and or the cabin(s), and the capability to produce additional pressurant in order to accommodate the repressurization gas.

Table 2.3.4.5.1.1.3-2 defines the pressurant requirements for both an airlock system and a depressurization system in support of EVA activities. Of primary concern with both systems are the weight penalties imposed in accomplishing this task. Figure 2.3.4.5.1.1.3-1 shows the required LO_2 and LN_2 volumes necessary to make up a depressurized volume. To minimize the impact of this requirement, the feasibility of using the stored cryogens that are carried as part of the propulsion system to provide pressurization gas were studied. Results show it is not necessary to carry separate LO_2 as the small volume make-up requirement can be meet by the primary LO_2 tanks. Based on the LN_2 volumes required, the study shows that the mass of the LN_2 storage tanks represents a major weight and volume savings over the use of an airlock.

Table 2.3.4.5.1.1.3-1: Airlock Mass/Volume Requirements

- **Current Planning for SSF Involves Two Interlocking Airlocks; One for Crew and one for equipment.**

Comparison of The SSF Airlock To LTS Crew Modules

	Mass (lbs)			Total Volume (ft^3)	Free Volume (ft^3)
	Structure	Support Equip	Total*		
Equip Airlock	3,150	7,100	10, 250	970	230
Crew Airlock	1,850	3,100	4,950	250	170
LTV Crew Module	—	—	13,640	1,307	900
LEV Crew Module	—	—	8,792	810	560

*Includes EVA Suits but not Crew

- **The Crew Portion of the SSF Airlock Is Equivalent to 36% of the Mass and 20% of the Volume of the LTV Crew Module, and to 56% of the Mass and 30% of the Volume of the LEV Crew Module.**

Table 2.3.4.5.1.1.3-2: Pressurant Make Up Requirements

Vehicle	Planned EVAs	Contingency EVAs	Backup	50% Safety Factor	Total
LTV	2*	2	2	3	9
LEV	2	2	2	3	9

*Could be Supplied by SSF

Vehicle	LN_2 Vol	LO_2 Vol **
LTV w/o Airlock	8.6 (ft^3)	1.8 (ft^3)
LTV with Airlock	1.6	0.36
LEV w/o Airlock	5.4	1.1
LEV with Airlock	1.6	0.36

LN_2 Tanks

Vol w/ 10% Ullage (ft^3)	Dia (ft)	LN_2 Wt (lbs)	Tank Wt (lbs)	Total Wt (lbs)
9.49	2.62	461.8	25.8	487.6
5.94	2.17	290	15.5	305.5
1.78	1.5	85.9	4.8	90.7

**Can be made up from main LO_2 reserves (propellant tanks)

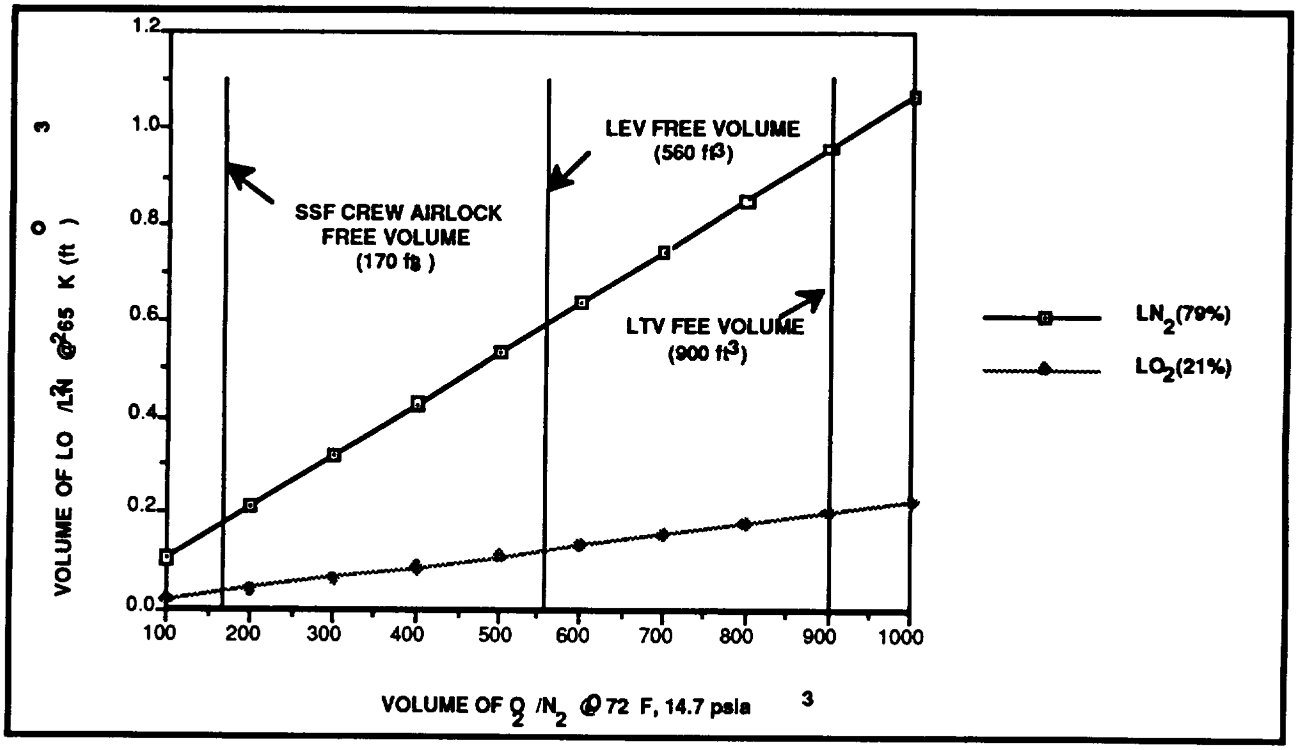


Figure 2.3.4.5.1.1.3-1: Volume of LO_2/LN_2 Required For Repressurization

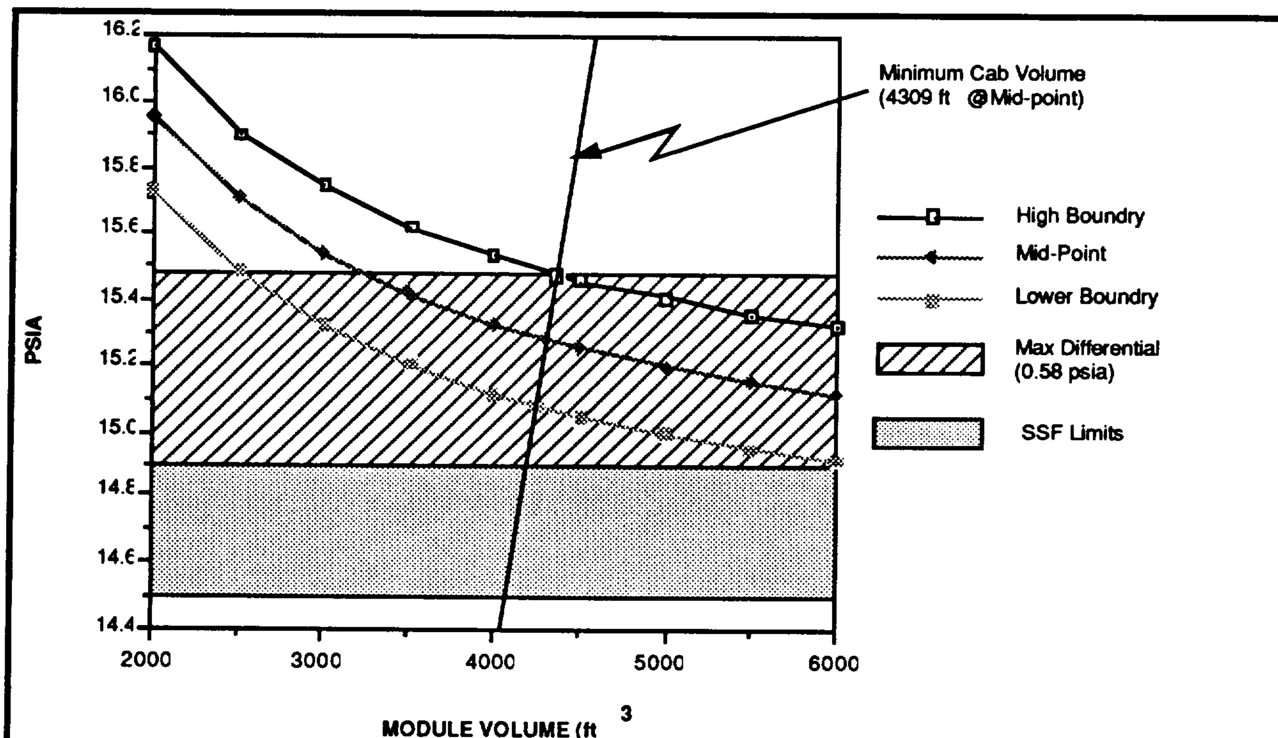


Figure 2.3.4.5.1.1.3-2 Cabin Volume and Pressure Relationship

During this feasibility study, an additional approach to using an airlock while minimizing the amount of pressurant required was evaluated. This approach returns pressurant within the airlock to the main cabin minimizing the amount of pressurant loss to leakage only. Further analysis indicated that although this approach appeared favorable initially, it failed to meet the maximum pressure differential of 0.58 psia that NASA-STD-3000 requires to prevent inner ear problems. Figure 2.3.4.5.1.1.3-2 shows the relation between cabin pressure and volume.

Summary and Recommendation - Comparison of the LTS crew module to the Apollo Command Module (CM) and the LEM was difficult as the mission requirements are drastically different. The preferred recommendation is new modules over modification of modules designed for different requirements. A derivation of the CM could be used as a crew rescue vehicle; although currently this is not an STV or SEI requirement. Based on this study the hybrid crew module concepts provide no advantage over either the single or separate module concept. The selection of a single module approach versus the two separate modules is dependent on the final LTS configuration. Separate modules are the recommended approach at this time if the LTS is made up of separate transfer and landing (excursion) vehicles; a single crew module is recommended for an LTS that employs a common transfer and landing vehicle. The weight and volume impact for implementing an airlock system in the crew module are extreme; however the entire module can be repressurized enough times to meet all EVA requirements for a minor weight penalty of 3.5% of the module mass. Therefore, our recommendation is that the cabin be depressurized then repressurized to support EVA activities.

2.3.4.5.2 ***Crew Visibility Analyses***—The objective of this study was to establish crew visibility requirements for the LTS , specifically for manual control of rendezvous , docking, and lunar descent and ascent. The key issues addressed were the manual override of autonomous functions involving rendezvous and docking and adequate crew visibility during lunar landing. Even though the LTS will be capable of complete automation, those operations involving rendezvous, docking, lunar descent, and landing require the crew to have direct window viewing of each of the operations.

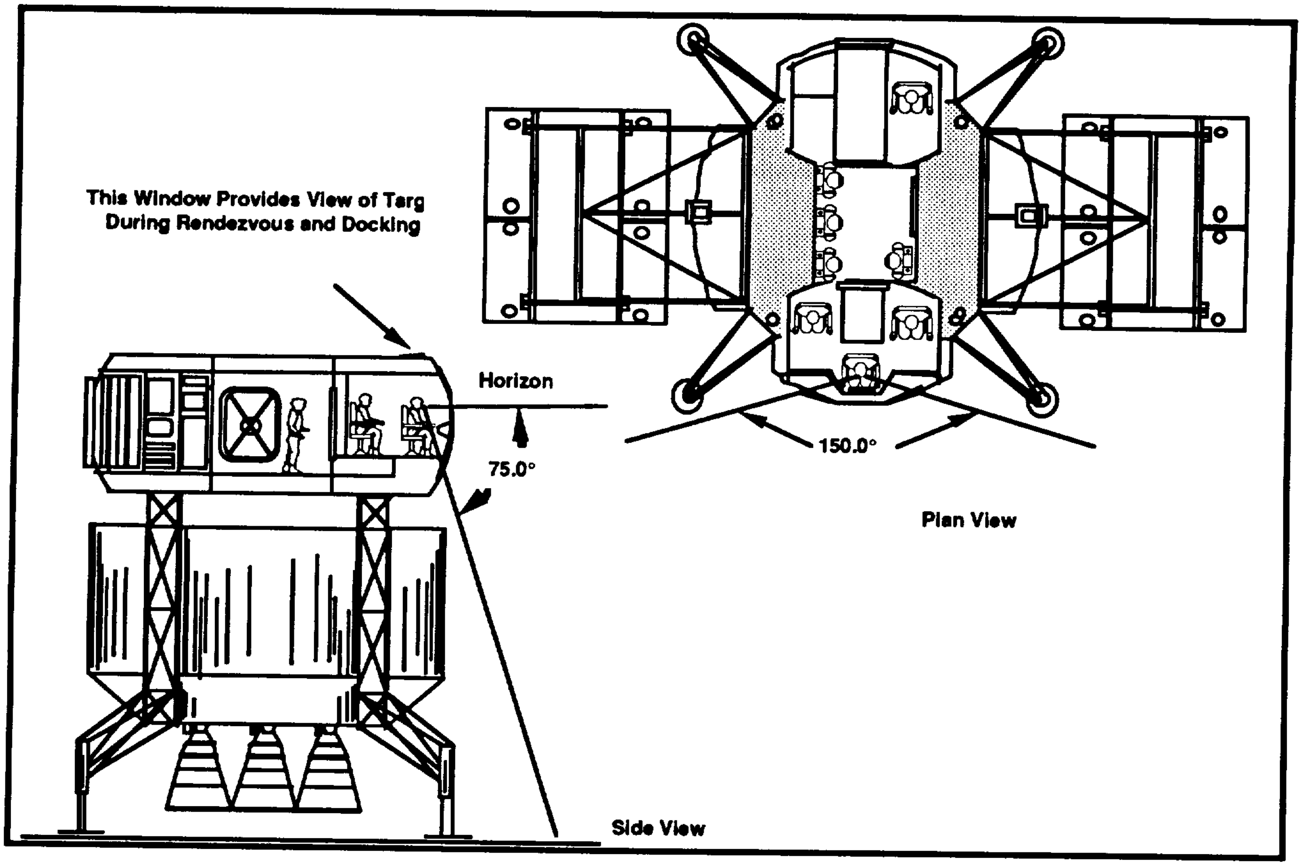
The groundrules and assumptions defined for this study included physical and production characteristics.

a) Astronauts shall be involved in docking.
b) Astronaut is the active and controlling element during lunar landing.
c) The crew module has windows and control and display consoles so that the crew can perform docking and lunar descent and ascent maneuvers.

d) The crew can override the automated rendezvous and docking system.

e) Automated rendezvous and docking in low lunar orbit (LLO) is provided for reusable cargo missions, whereas piloted missions provide crew monitoring and control to rendezvous and docking.

f) During landing operations, the crew module provides two crew members with console positions and windows to visually monitor all critical landing activities, including forward landing pad touchdown.



Analysis - The methodology used for this study incorporates lessons learned from the Apollo program and the current NASA philosophy which has the crew fully involved in all in-flight operations. The Apollo program demonstrated that the pilot required full visual view of the target during the final phase of rendezvous and docking, as well as being positioned as close to the viewing window as possible to provide adequate field-of-vision (FOV) for landing. The analysis of this study was based on maintaining LEM design criteria for FOV, since the system has been proven. This criteria consisted of a downward angle of 65 degrees and 160 degrees across the horizontal plane. One drawback that the LEM encountered, and is anticipated to exist within the LTS, is the ability to see backwards in support of final landing maneuver corrections. This visibility issue should be resolvable through the use of video displays located on the control console.

Summary and Recommendations - The design of the crew module will incorporate the appropriate number of windows for viewing all critical operations. Every effort will be expended to assure adequate window viewing to provide as large a FOV as possible. Figure 2.3.4.5.1.2.2-1 shows the current crew module configuration and the available FOV in both the vertical and horizontal planes, windows have also been provided allowing the crew to observe the rendezvous and docking operation in LLO.

3.0 STV CONCEPT DEFINITION

The STV family of vehicles that came out of the Concept Selection Trade Study analysis shows that the lunar missions impose the most stringent requirements on the STV. The design approach has been to develop a vehicle that meets the design requirements and then evaluates the design to identify the elements that best satisfy the mission requirements for an initial ground-based STV, a later space-basing of the STV and, finally, the Mars mission profile.

The STV concept definition for a lunar mission vehicle is based on the requirements set out in the STV Statement of Work (SOW), with additional derived requirements from the Option 5 Planetary Surface System (PSS) documents, and the system trade studies and analyses. These studies and analyses recommend that the orbital mechanics designated as Lunar Architecture #1 (LA#1) is the best at meeting these requirements. Briefly stated, LA#1 uses a LEO node as the start and finish of the lunar mission for both crew and cargo flights. The LEO node is used for assembly, checkout, and refurbishment. Additional elements of the orbital mechanics require the vehicle orbit in low lunar orbit (LLO) before lunar descent, have a lunar trajectory that encompasses a free earth return for an abort scenario, and returns to the LEO node via an aerobraking pass through the atmosphere.

Once the lunar mission profile shown in Figure 3.0-1 was selected, the following key design drivers, as identified in Section 2.2.2, were integrated into the development and definition of vehicle configuration candidates.

a) The system shall deliver 14.6 tonnes of cargo and 4 crew to lunar surface and return
b) The system shall deliver 33.0 t of cargo on an unmanned flight to the lunar surface
c) LEO transportation node shall be Space Station Freedom (SSF)
d) The propulsion system shall use cryogenic propellant
e) The system shall be reusable for a minimum of five missions

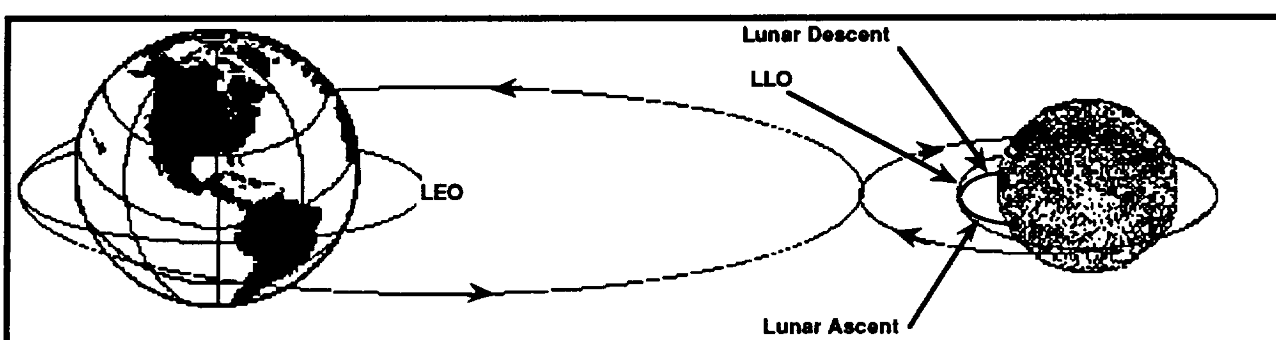


Figure 3.0-1: Lunar Mission Profile

These design drivers were also filtered through the subsystems trade study analysis (Section 2.3.4) and finally incorporated into the vehicle design. (Note: The trade study analyses are documented in a later section and only the results are shown in the concept definition as they apply to the overall vehicle definition.)

3.1 LUNAR STV CONCEPT DEFINITION

The STV consists of a family of vehicles which share common elements performing both cargo and piloted/cargo missions such as GEO delivery, lunar, and planetary (Mars mission). That portion of the STV family that deals with the lunar missions is called the lunar STV or the Lunar Transportation System (LTS). The LTS is comprised of two mission profiles - (1) the cargo mission, capable of delivering 33 tonnes to the lunar surface and (2) the piloted/cargo mission, capable of delivering a crew of 4 plus 14.6 tonnes to the lunar surface. These mission profiles reflect the flights and cargo manifesting schedules developed from the Option 5 Lunar Exploration Requirements Levels I - III.

A derived requirement was developed from the studies that the final cargo and piloted vehicles would share common elements, producing a family of vehicles that have common structural core, propulsion and avionics equipment, drop tanks, and can be configured for either type of mission with no major modification to these elements. The detail definition of each vehicle configuration, performance, mass properties, structure, subsystem, cargo and crew handling, and operations will be discussed in the following section. The evolutionary aspects of the configuration to perform the initial STV mission and the planetary mission are detailed at the end of this section.

3.1.1 Piloted Concept Overview

The LTS piloted configuration for the single propulsion system concept is shown in Figure 3.1.1-1. A crew module, six drop tanksets, and an aerobrake with its associated equipment are added to the propulsion/avionics core. The piloted vehicle dry mass is 27.58 tonnes. The configuration can deliver 15.26 tonnes of cargo (14.6 tonnes cargo plus cargo supports) in addition to the crew of 4 to the lunar surface and return the vehicle and crew to LEO using approximately 174 tonnes of LO_2/LH_2 propellant. TEI and LOI propellant is housed in the drop tank sets, ascent and descent propellant is found in the core, and the return propellant is housed in two sets of tanks within the aerobrake. The 13.72 m rigid aerobrake has been designed to protect the crew during the aeroassisted maneuver before returning to Space Station Freedom.

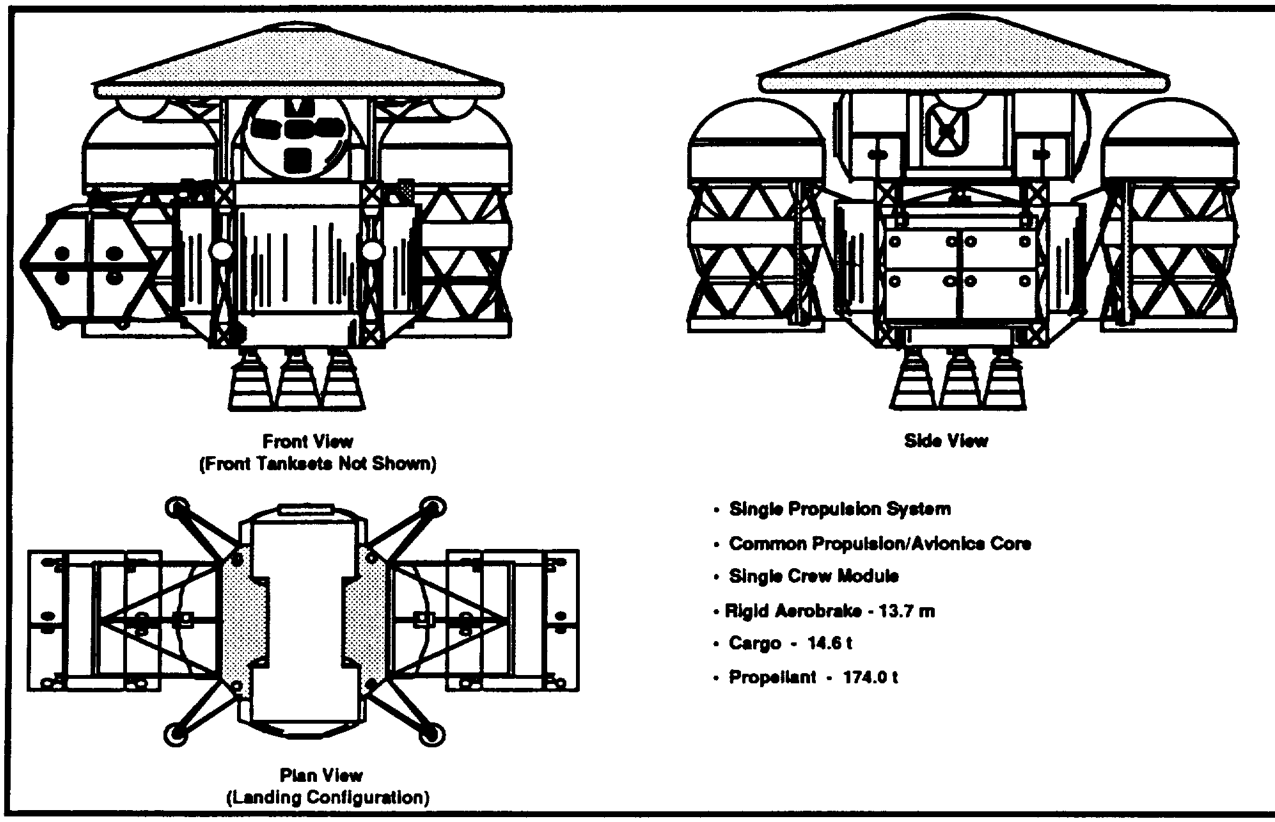


Figure 3.1.1-1: Piloted LTS Configuration

3.1.2 Cargo Concept Overview

The LTV cargo expendable configuration for the single propulsion system concept is shown in Figure 3.1.2-1. To form the cargo expendable configuration, a cargo platform (10.5 m x 14.8 m) and six drop tanksets have been added to the propulsion/avionics core. The cargo vehicle dry mass is 18.75 tonnes and can deliver 33 tonnes of cargo to the lunar surface using 146.5 tonnes of LO_2/LH_2 propellant loaded into the drop tanks and core tanks. The Flight 1 cargo manifest shown in the plan view is a typical arrangement for the four cargo missions.

3.1.3 Performance Overview

There are three missions designed for the LTS: piloted, cargo expendable and an optional cargo reusable. Vehicles were sized and capabilities, propellant loads, and IMLEOs determined based on the cargo requirements and the groundrules established for the STV study. The piloted mission (crew plus 14.6 tonnes of cargo) was found to be vehicle sizing driver. Once the baseline vehicle was determined, the cargo capabilities shown in Table 3.1.3-1 defined a maximum capability for an expendable cargo mission of 37.4 tonnes, or 4.4 tonnes over the required capability. The

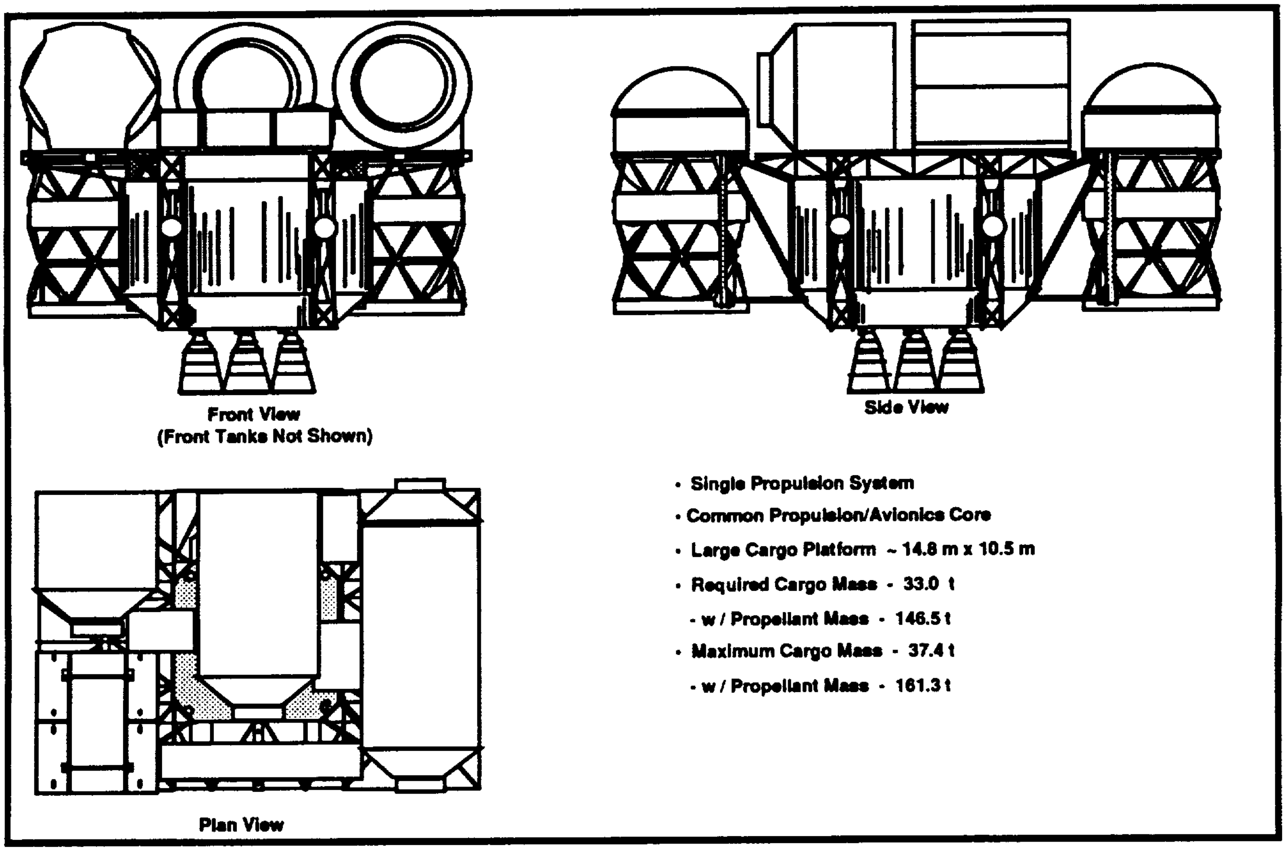
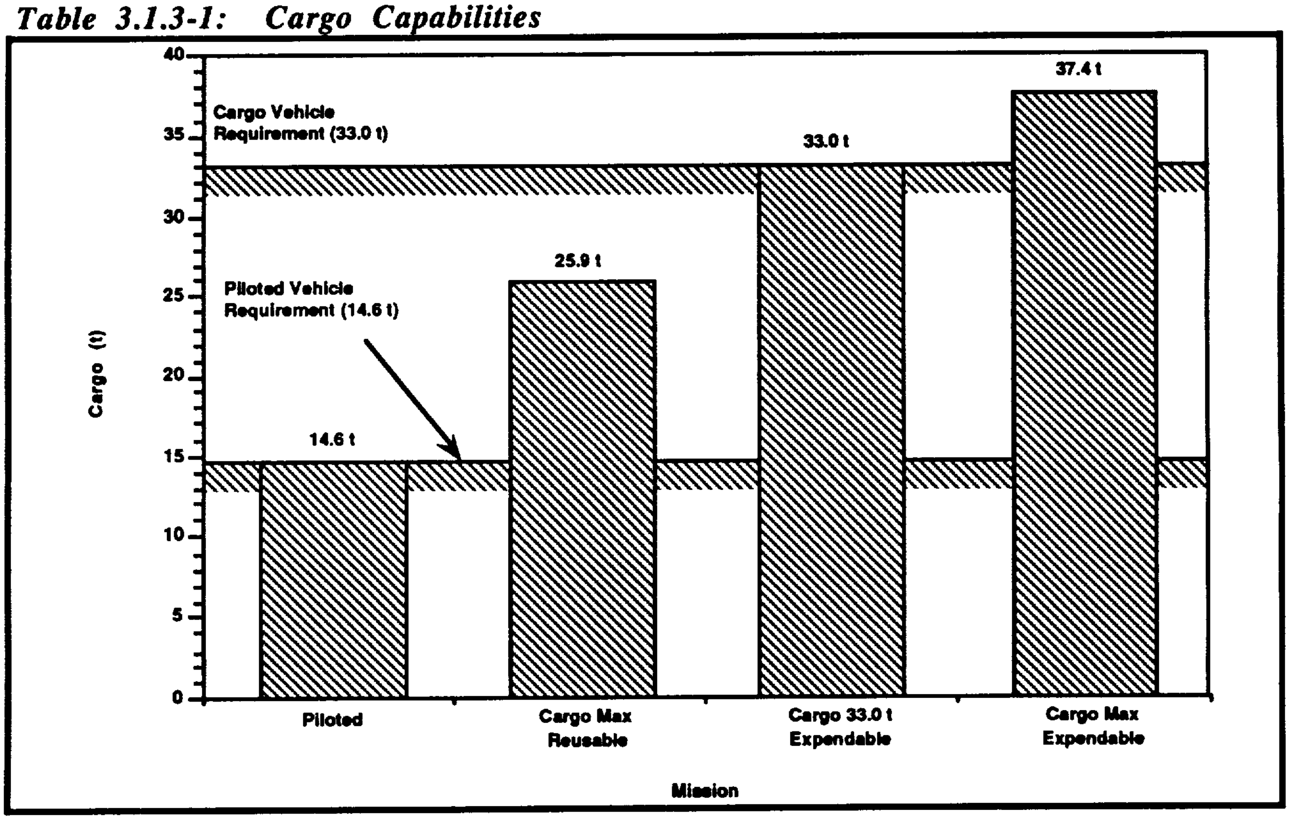


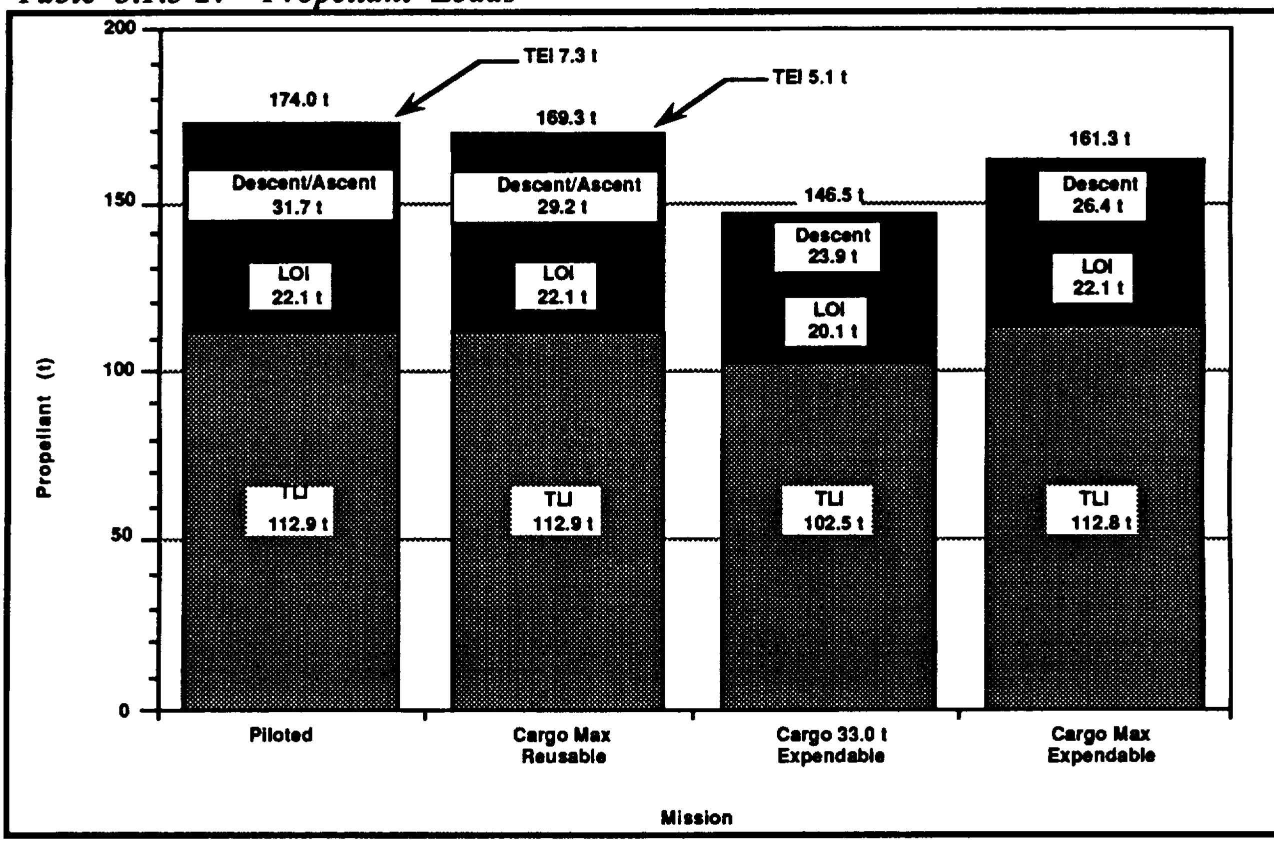
Figure 3.1.2-1: Cargo LTS Configuration

Table 3.1.3-1: Cargo Capabilities



required delivery of 33 tonnes of cargo is met by offloading 27.5 tonnes of propellant. The optional cargo reusable mission deliver 25.9 tonnes of cargo with a full propellant load and return to SSF.

Table 3.1.3-2: Propellant Loads



3.2 SUBSYSTEM COMMON ELEMENTS

The common propulsion/avionics core shown in Figure 3.2-1, represents the heart of the single propulsion system family vehicle. Crew module, aerobrake, cargo pallets or platforms, and drop tanksets can be added to form various configurations allowing the STV vehicle family the versatility to capture other missions. The core consists of five internal propellant tanks (4 LH_2 and 1 LO_2 tanks), primary structure, the four landing legs mounted to the lower cross beam, and critical subsystems - the propulsion system of five Advanced Space Engines (ASE), RCS, GN&C, communication & data handling, power, and thermal control. Table 3.2-1 provides the core vehicle mass properties breakdown of these systems.

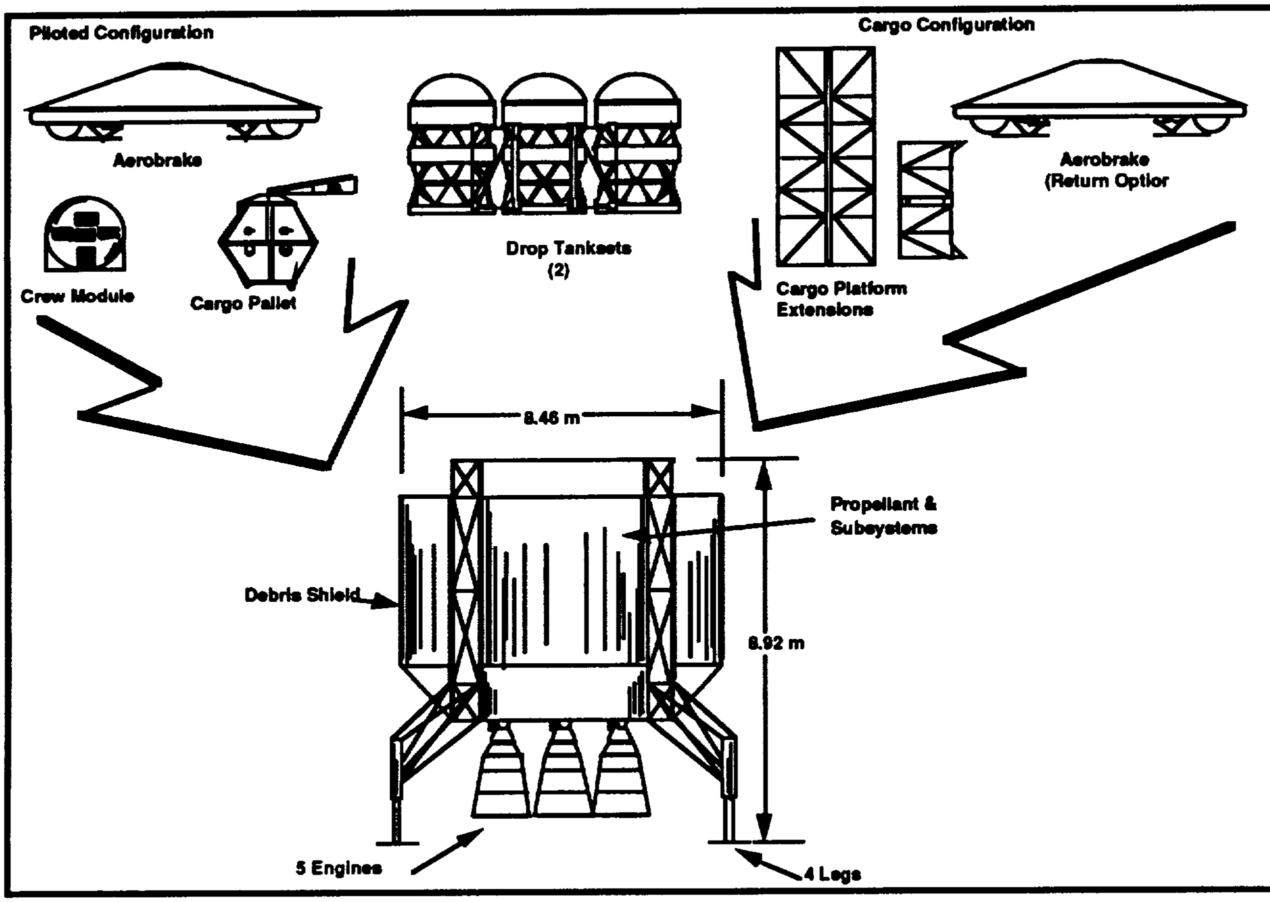


Figure 3.2-1: Propulsion/Avionics Core Module

Table 3.2-1 Mass Properties Breakdown - Core Vehicle

	DESCRIPTION	MASS	MASS
	CORE VEHICLE SUMMARY	KG	M.TONS
02	STRUCTURE	2363.15	2.36
03	PROPELLANT TANKS	802.86	0.80
04	PROPULSION SYSTEM	380.34	0.38
05	MAIN ENGINES	1150.11	1.15
06	RCS SYSTEM	122.45	0.12
07	G. N. & C.	195.46	0.20
08	COMMUNICATION & DATA HNDLG	242.70	0.24
09	ELECTRICAL POWER	444.22	0.44
10	THERMAL CONTROL SYSTEM	553.47	0.55
11	AEROBRAKE	0.00	0.00
19	GROWTH	938.21	0.94
	DRY WEIGHT	7192.97	7.19

3.2.1 Structure

The following section deals with the structural elements of the propulsion/avionics core. The elements include the airframe, core and drop tank sizes, material and mass, meteoroid and debris shielding, and the general arrangement of the equipment located in the core. The meteoroid and debris shielding sizing requirements are discussed in another section of the report.

3.2.1.1 Core Structure Details

The propulsion/avionics core primary structure is composed of graphite epoxy square tubing with aluminum end fittings forming two trusses consisting of a lower and upper box beam and the connecting longitudinal members. The lower cross beam is the thrust frame, equipment mount and support structure for the landing legs. The upper cross beam supports the cargo platform, crew module and payloads.

The secondary structural members are graphite epoxy round tubing with aluminum end fittings. They tie the two trusses together and form the mounting braces for the four LH_2 tanks. Figure 3.2.1.1-1 gives an overview of the major core structure.

The structural dimension details are shown in Figure 3.2.1.1-2, views A & B. The overall height of the structure is 8.92 m with landing legs deployed. The landing legs are spaced 12.2 m apart and extend 2.74 m below the basic airframe. The primary frame is a 5.64 m square and 5.25 m high. The depth of the primary structure top and bottom truss work is 0.9 m, with the engine mounts located approximately 2 m from the centerline of the core. Plan View and View C provide additional detailed structural dimension for the propulsion/avionics core and the supports for mounting the drop tanksets. Addition of this drop tankset structure increases the core to 8.46 m in diameter. The entire structure is cover with a graphite polyimide debris shield.

3.2.1.2 Core Tanks

The isometric view of the propulsion/avionics core shown in Figure 3.2.1.2-1 (Figure 7), locates the five core tanks - 4 LH_2 tanks and 1 LO_2 tank. The spacing between the tanks and the structure is used for packaging the subsystem components. Graphite polyimide debris shields are attached to the four sides of the core structure to provide micrometeoroid and debris protection for the tanks.

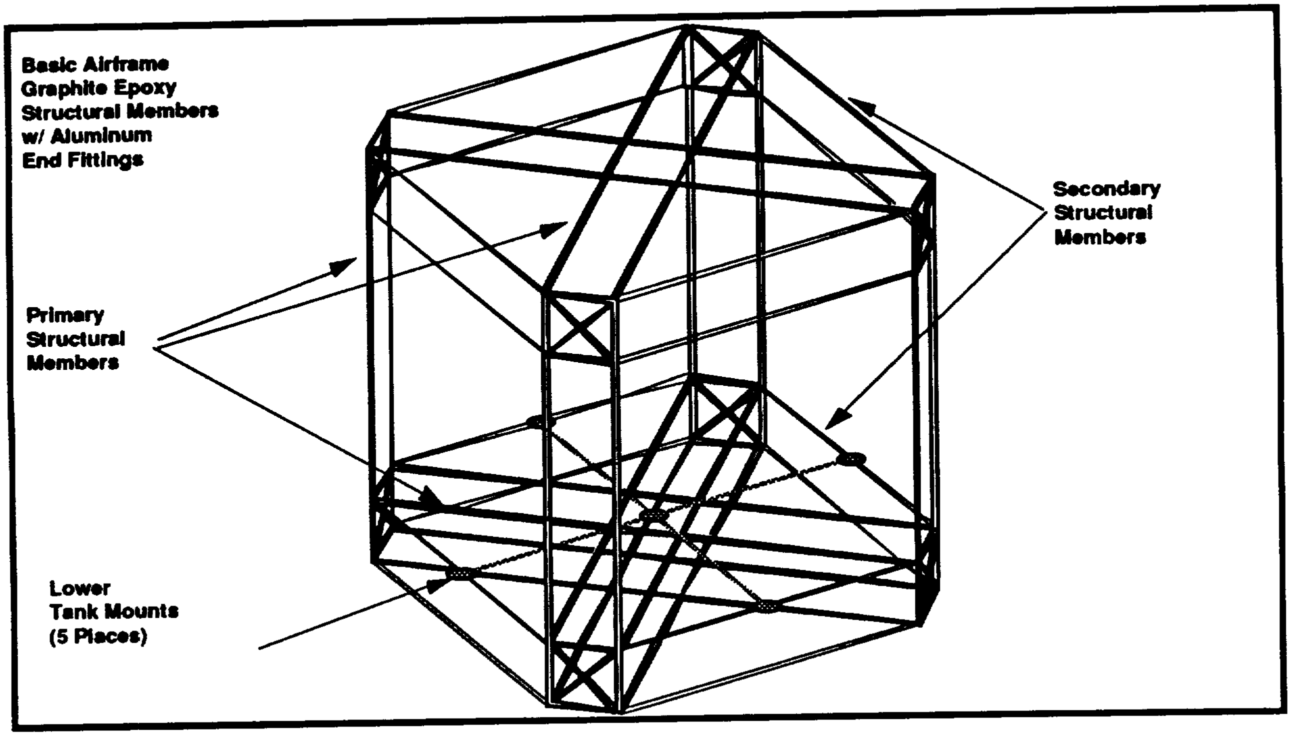


Figure 3.2.1.1-1: Overview of the Major Core Structure.

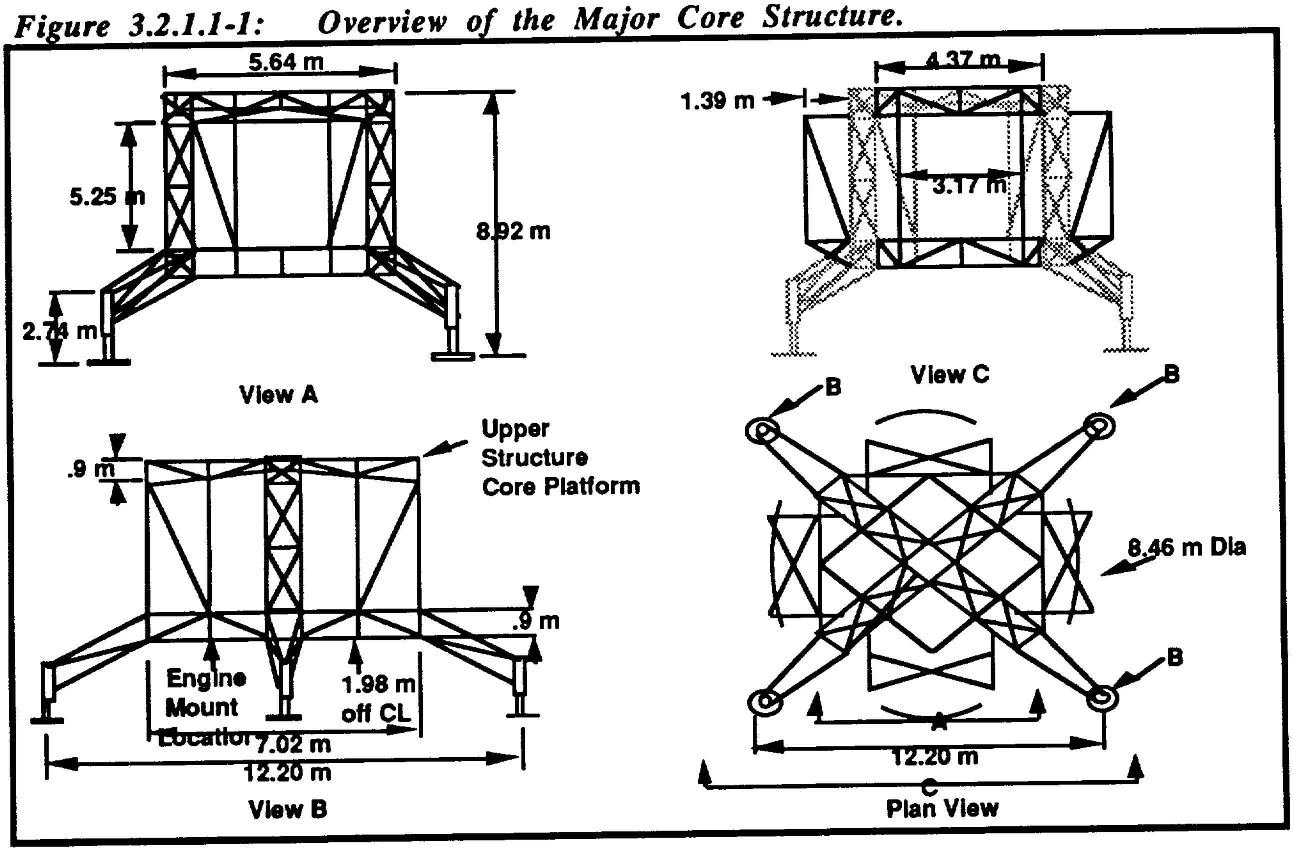


Figure 3.2.1.1-2: Detailed Core Module Structural Dimension

The details of the propulsion/avionics core tanks are shown in Figure 3.2.1.2-2. The four LH_2 tanks, composed of aluminum-lithium spun domes and isogrid barrel panels to conserve weight, are spaced symmetrically around a center LO_2 tank and mounted to the upper and lower cross beams of the core structure. The LO_2 tank is 4.4 m in length and 2.9 m in diameter, the LH_2 tanks are 4.2 m long and 2.6 m diameter. Combined, these tanks represent a total propellant capacity of 32.5 tonnes.

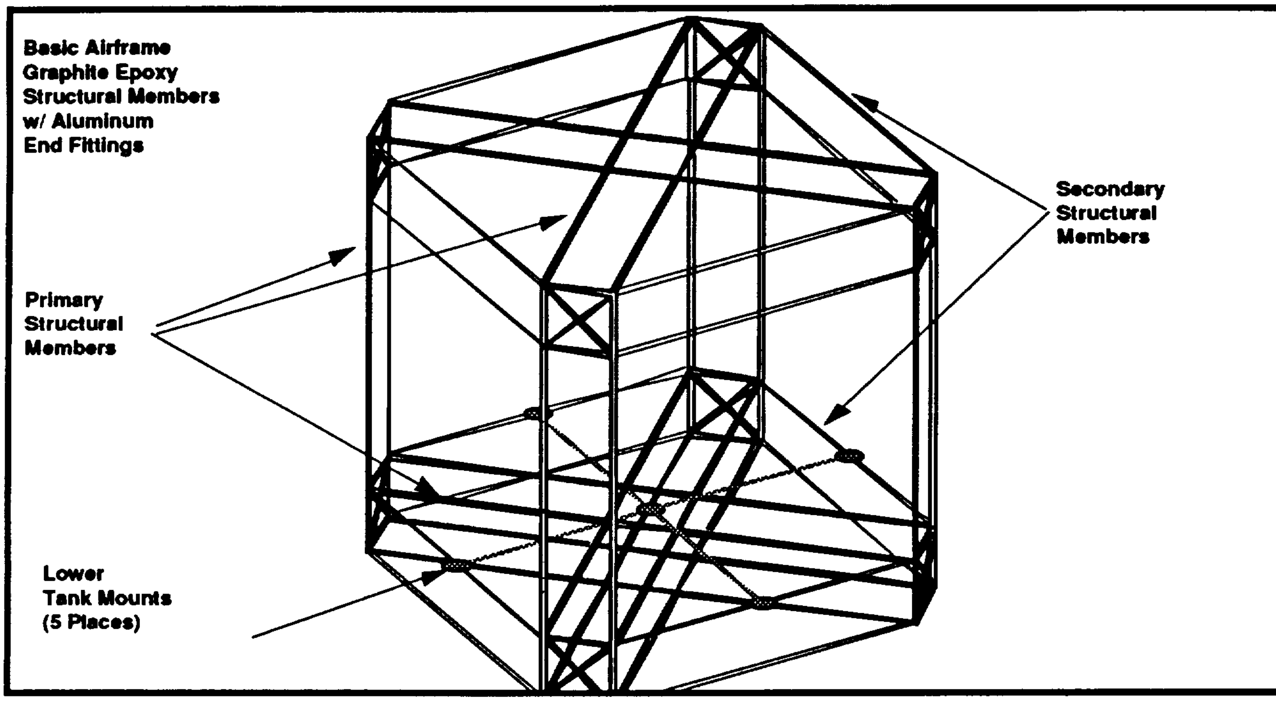


Figure 3.2.1.2-1: Isometric View of the Propulsion/Avionics Core

3.2.1.3 Equipment Layout

Figure 3.2.1.3-1 shows the packaging arrangement of the propulsion/avionics core equipment. The placement and size of the propellant tanks allow the subsystem equipment to be packaged in spaces created between the trusses and the tanks (Figure 3.2.1.2-2 - Plan View). The various tanks for potable water, helium, GO2, and GH2 are packaged in two of the four bays with the fuel cells occupying the other two and the avionics equipment bays located in the space formed by the upper cross beams. By packaging this equipment around the top and sides of the vehicle, access for repair or change out is possible. Leg deployment mechanisms are placed in the lower portion of the core structure and docking ports are provided in the top of the core.

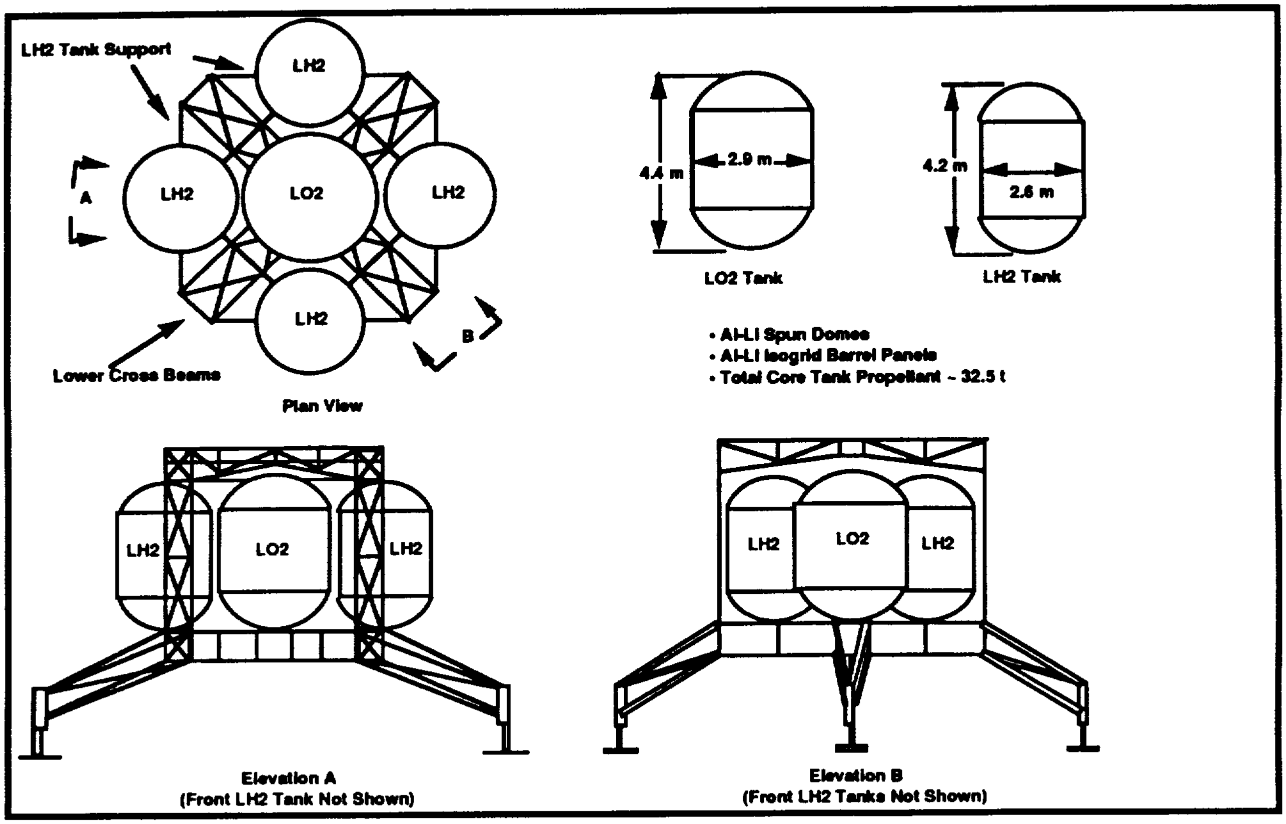


Figure 3.2.1.2-2: Details of the Propulsion/Avionics Core Tanks

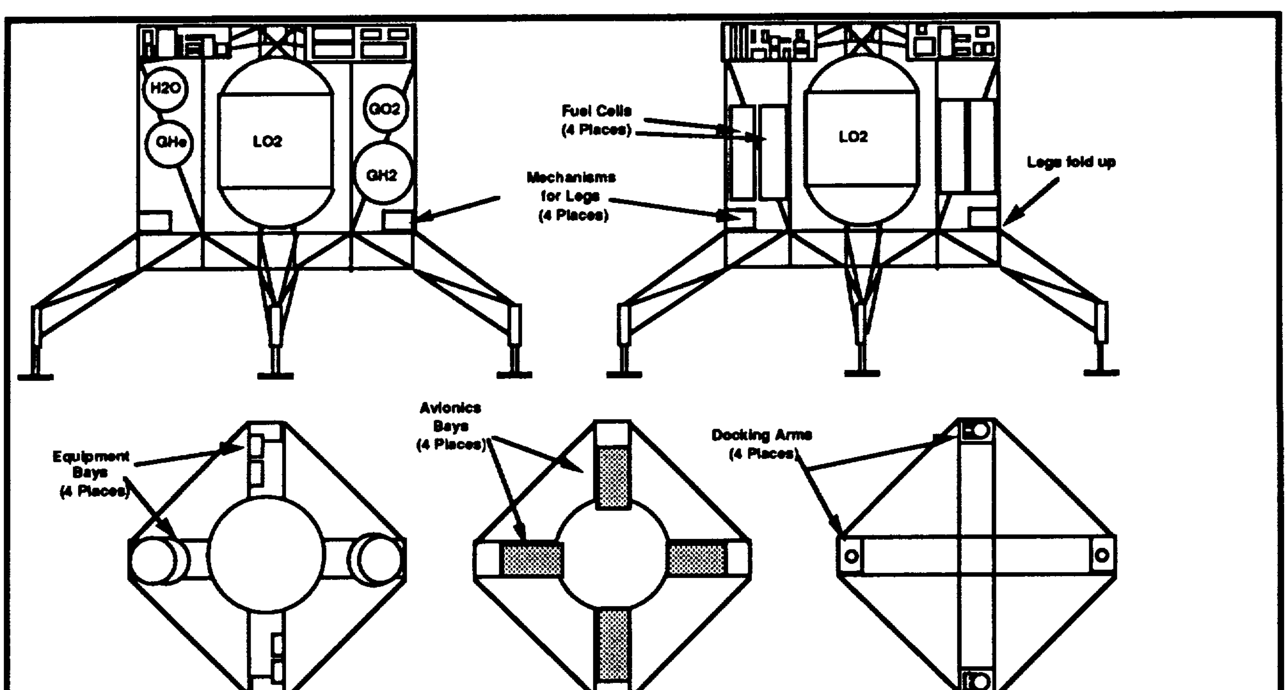


Figure 3.2.1.3-1: Packaging Arrangement for the Propulsion/Avionics Core Equipment

3.2.1.4 Drop Tanks

The LTS carries two tank arrangements, one on each side of the vehicle, each consisting of three drop tanksets (2 TLI and 1 LOI). Figure 3.2.1.4-1 shows the details of a typical tank arrangement. The propellant capacity of an individual drop tankset consisting of 1 LH2 tank and 1 LO2 tank is approximately 28 tonnes, and when combined into a tank arrangement 84 tonnes. Tanks are constructed of aluminum-lithium domes and isogrid barrel panels Weight saving graphite-epoxy struts and frames connect the tanks. Support structure connects them to the adjoining tankset and to the core vehicle. The tanks fit within a 4.6 m (15 ft) payload shroud. For ground heat leaks and on orbit thermal protection, tanksets have spray-on-foam-insulation (SOFI) and multi-layer insulation (MLI). A helium pressurization system and special instrumentation for monitoring are integrated into each tankset . Mass properties of the tanksets are shown in Table 3.2.1.4-1.

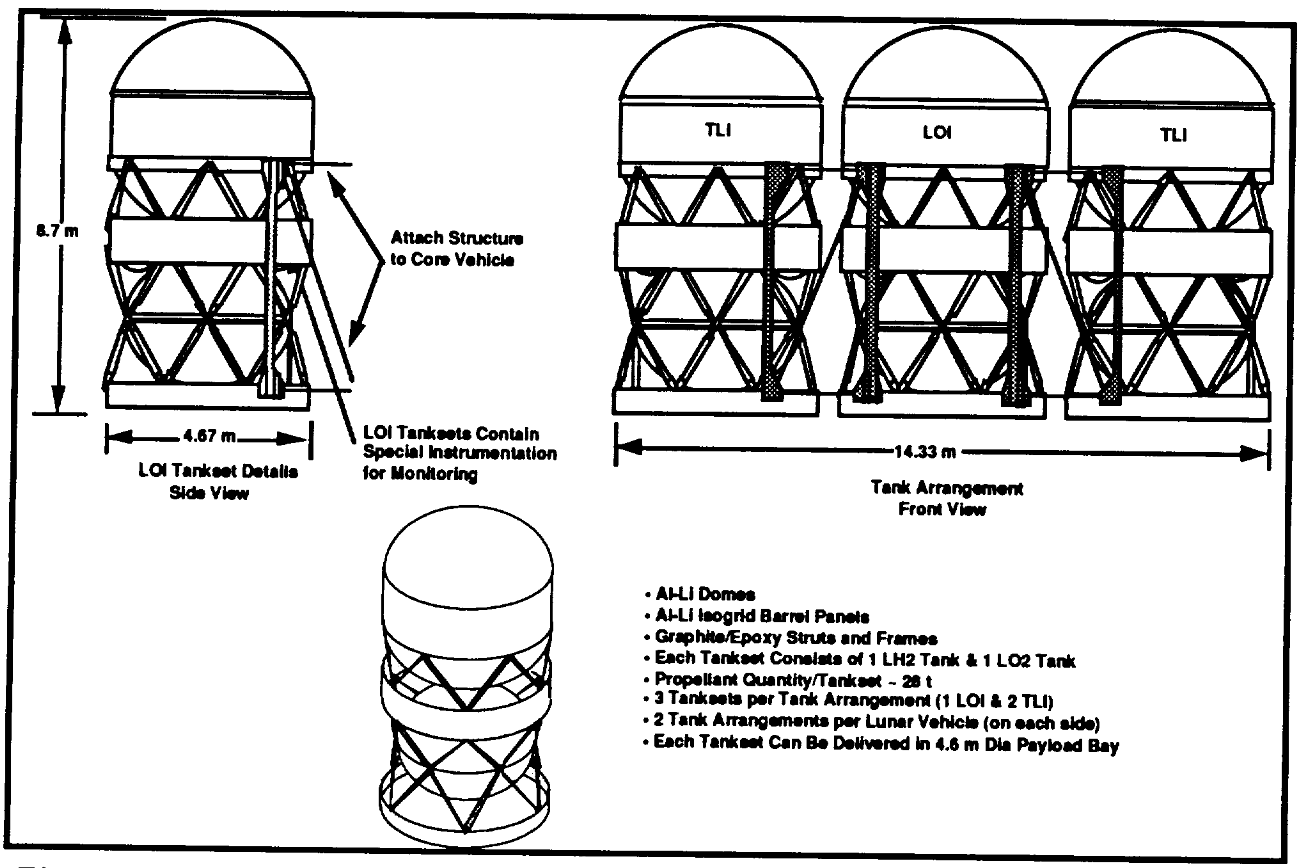


Figure 3.2.1.4-1: Typical Tank Arrangement Details

Table 3.2.1.4-1 Mass Properties Summary TLI/LOI Tanks

	Mass Properties Tanksets	
Components	LOI (Mass t)	TLI (Mass t)
LH2 Tank	0.34	0.34
LO2 Tank	0.18	0.18
Structure	0.07	0.07
TPS	0.42	0.42
Subsystems	0.46	0.23
Contingency (15%)	0.22	0.19
Dry Mass	1.69	1.43

The mounting of the drop tank arrangements to the propulsion/avionics core is shown in Figure 3.2.1.4-2. The two TLI tanksets attach to the center LOI tankset using struts with end fittings using clip-in locking pins. The LOI tankset is directly mounted to the core structure using a similar strut and end fitting arrangement.

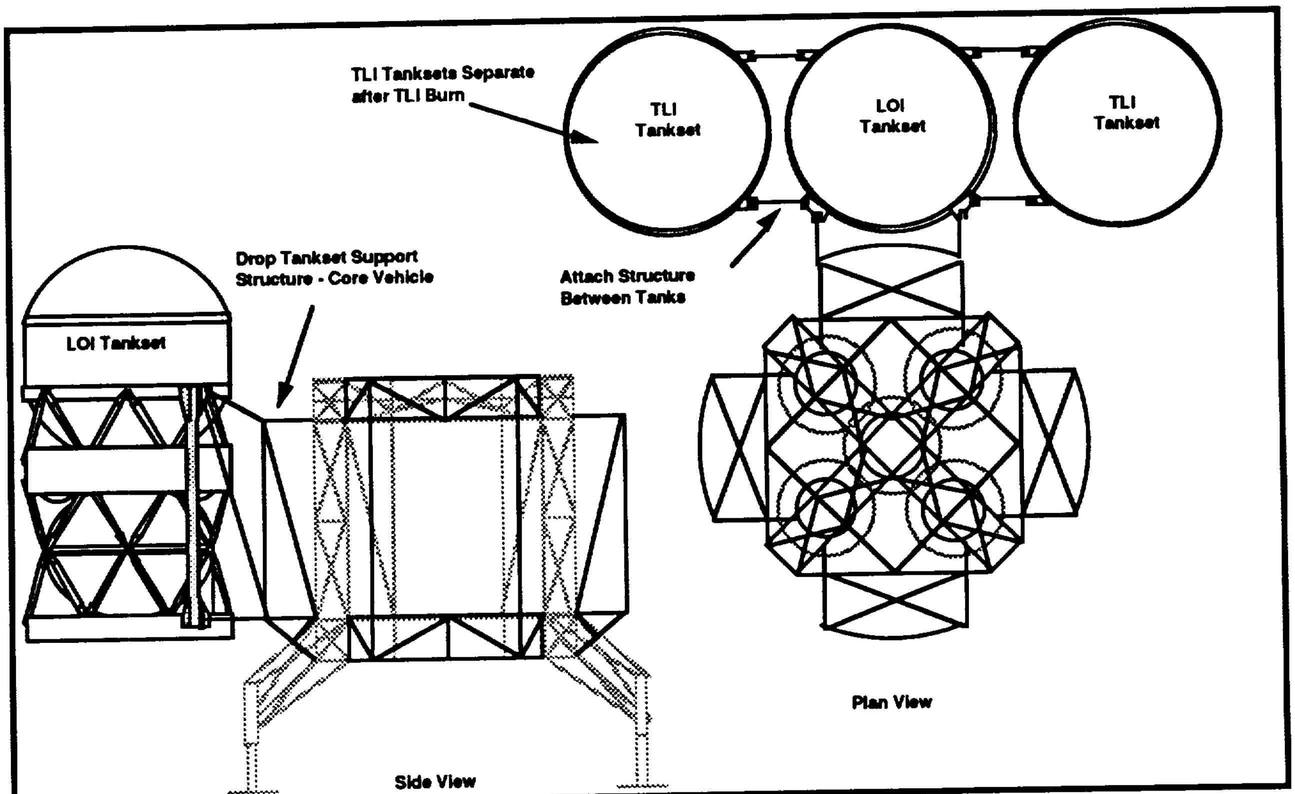


Figure 3.2.1.4-2: Drop Tank Arrangements Mounted To The Propulsion/Avionics Core

The positioning of the TLI tanksets allows them to be separated after the TLI burn. The LOI tankset remains with the vehicle until LLO insertion, when it is then released.

3.2.2 Propulsion System

This section describes the propulsion/avionics core propulsion subsystem that consists of: the main engine system, RCS system, a propellant management system, propellant tanks and their associated feed lines.

3.2.2.1 Engines

The layout of the main propulsion engines is shown in Figure 3.2.2.1-1. Five advanced space engines are mounted to the lower cross beams of the core, spaced 2 meters from center to center of engines, with a nozzle exit diameter of 1.34 meters. This spacing pattern accommodates a gimbal range of ± 8° except for the center engine which is not required to gimbal. Electrical mechanical actuators are used for gimbaling.

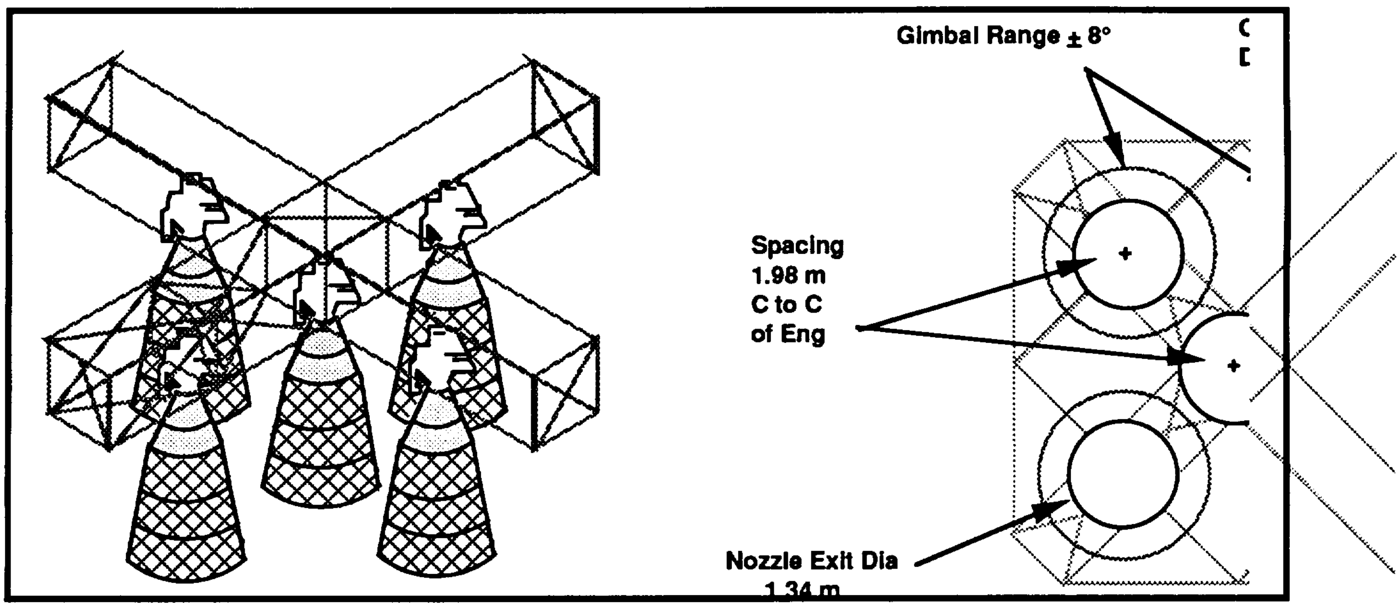


Figure 3.2.2.1-1 Main Propulsion Engine Layout

Attachment of the engines to the core occurs through vehicle/engine carrier plate quick disconnects, allowing easy change out during maintenance. The vehicle carrier plates are incorporated into the lower portion of the box beam engine support. The engine is assembled onto an engine carrier plate including all of the engine interfaces, which is then mated with the vehicle carrier plate disconnects, as shown in Figure 3.2.2.1-2. Additional details of the engine carrier plate are shown in Figure 3.2.2.1-3. The disconnects shown penetrate the vehicle carrier plate and lock into place to complete installation of the engine. A common engine interface approach was used to allow different engine versions to be installed as upgrades are made or for the tailoring of the engine configuration to specific missions.

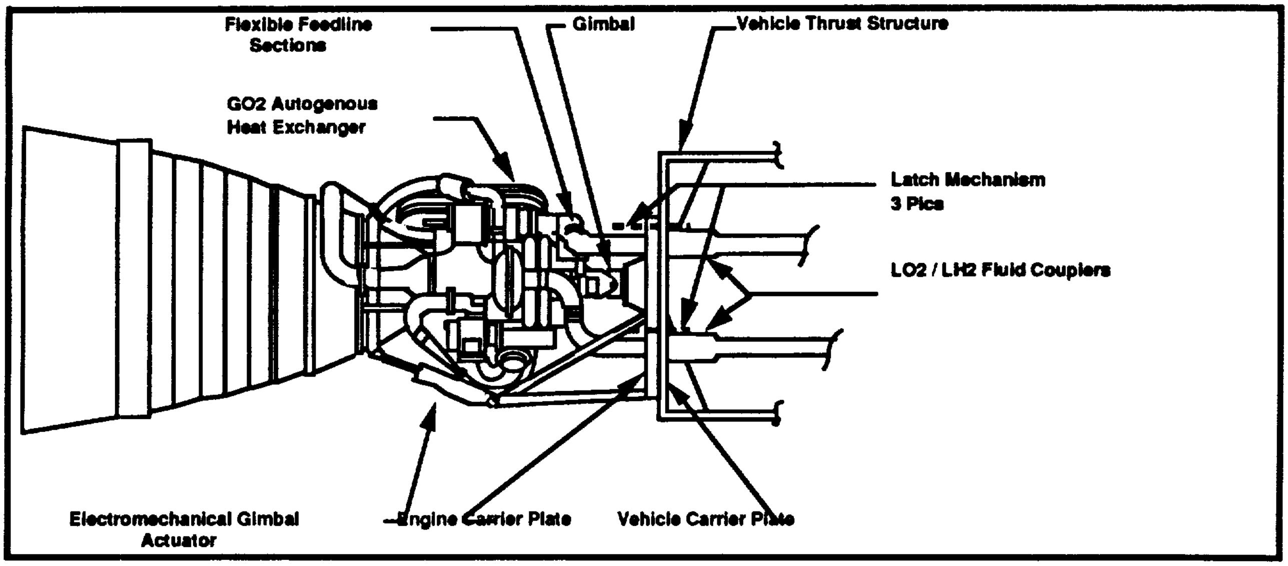


Figure 3.2.2.1-2 Engine Replacement

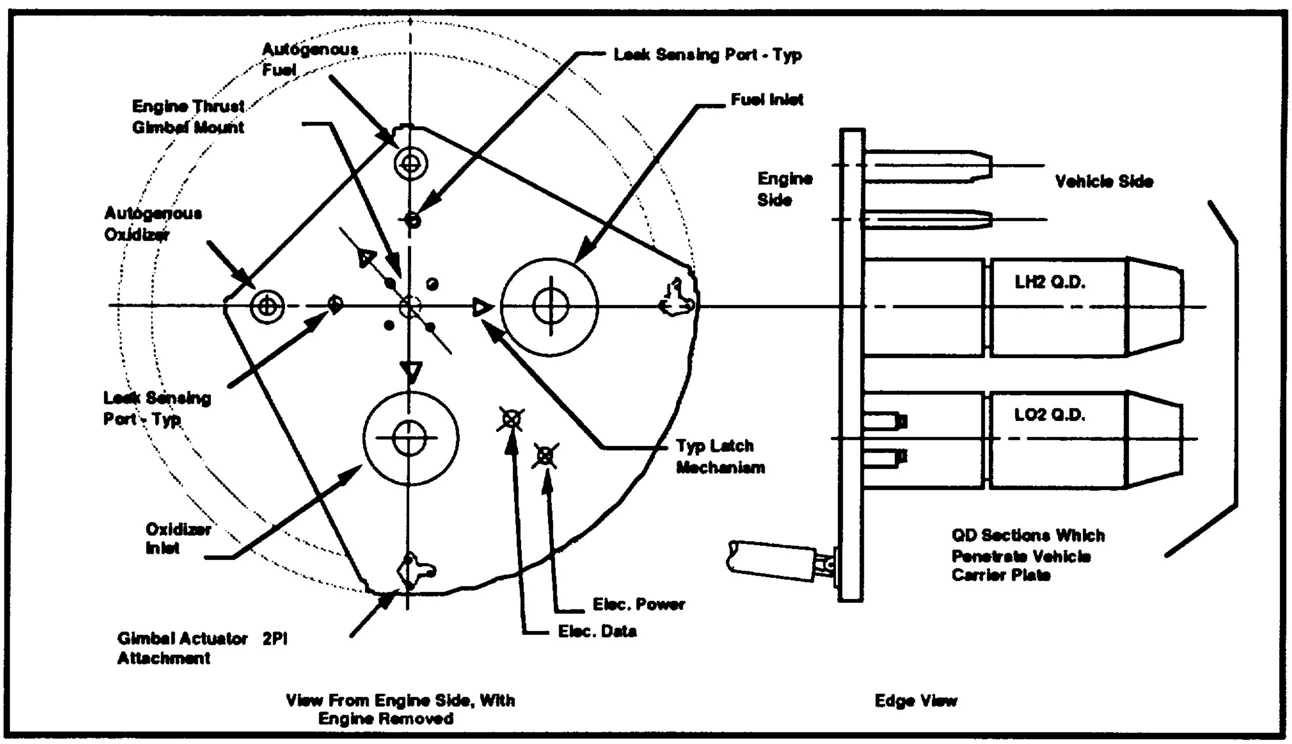


Figure 3.2.2.1-3 Engine Carrier Plate

3.2.2.2 Reaction Control System (RCS)

The LTS RCS thrusters consist of two separate systems as shown in Figure 3.2.2.2-1, one located on the propulsion/avionics core and the other on the aerobrake. As shown, six degrees of freedom are provide by 24 variable thrusters. Analysis recommended that the core vehicle RCS system be self contained and kept off the cargo and crew module, even though more leverage would be available. The thrusters at the upper end of the core vehicle are inactive when fully assembled.

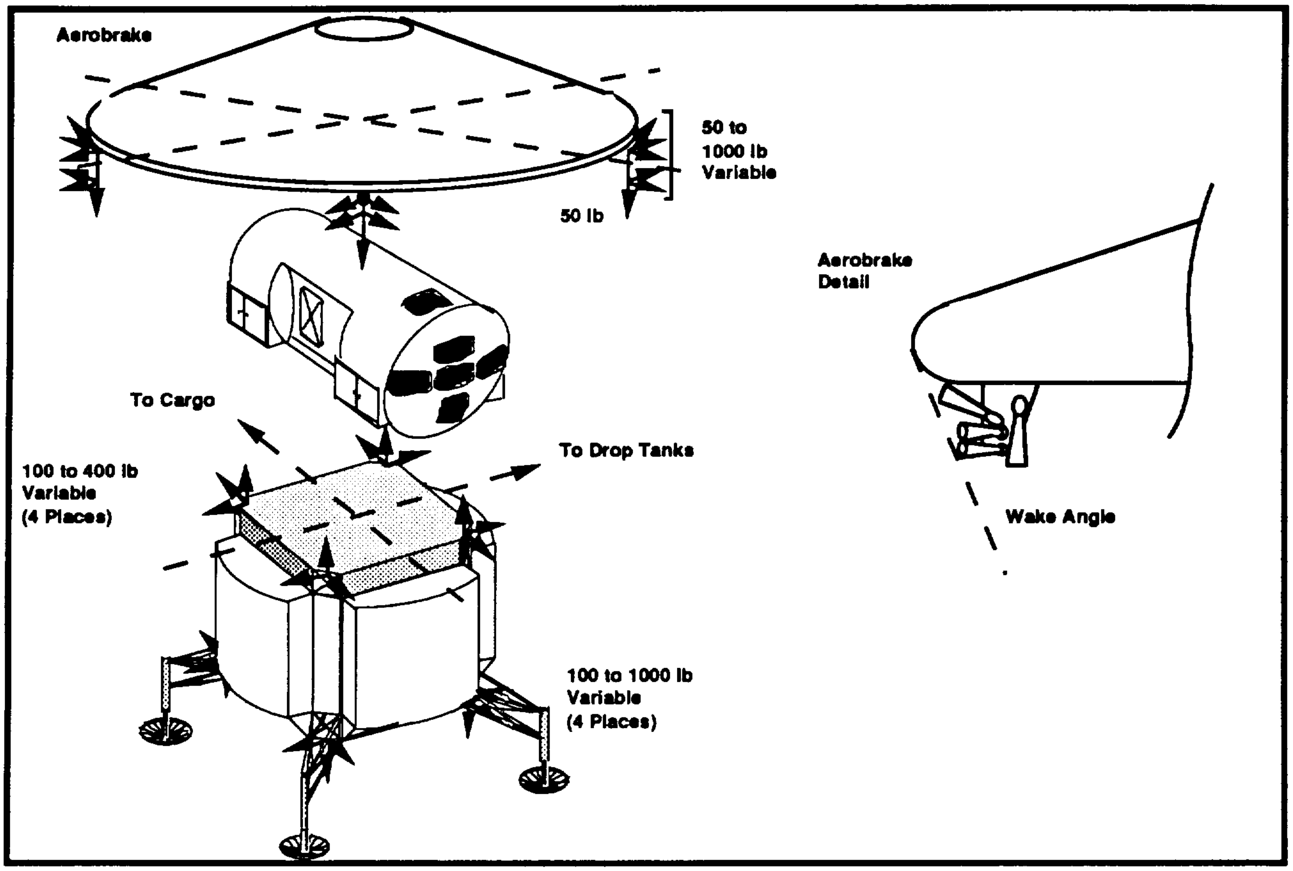


Figure 3.2.2.2-1 RCS Thruster Arrangement

3.2.2.3 Drop Tank Feed Lines and Disconnect

The feed lines shown in Figure 3.2.2.3-1 connect the two TLI tanksets (both LO2 and LH2) through an umbilical to the LOI tankset that then merges at an umbilical connection to the core tanks. When the TLI tanksets are separated after TLI burn, the propellant disconnect is made at this TLI/LOI umbilical, with the LOI disconnect made at the LOI/core tank umbilical. Figure 3.2.2.3-2 gives a typical fluid schematic for each of the tanksets.

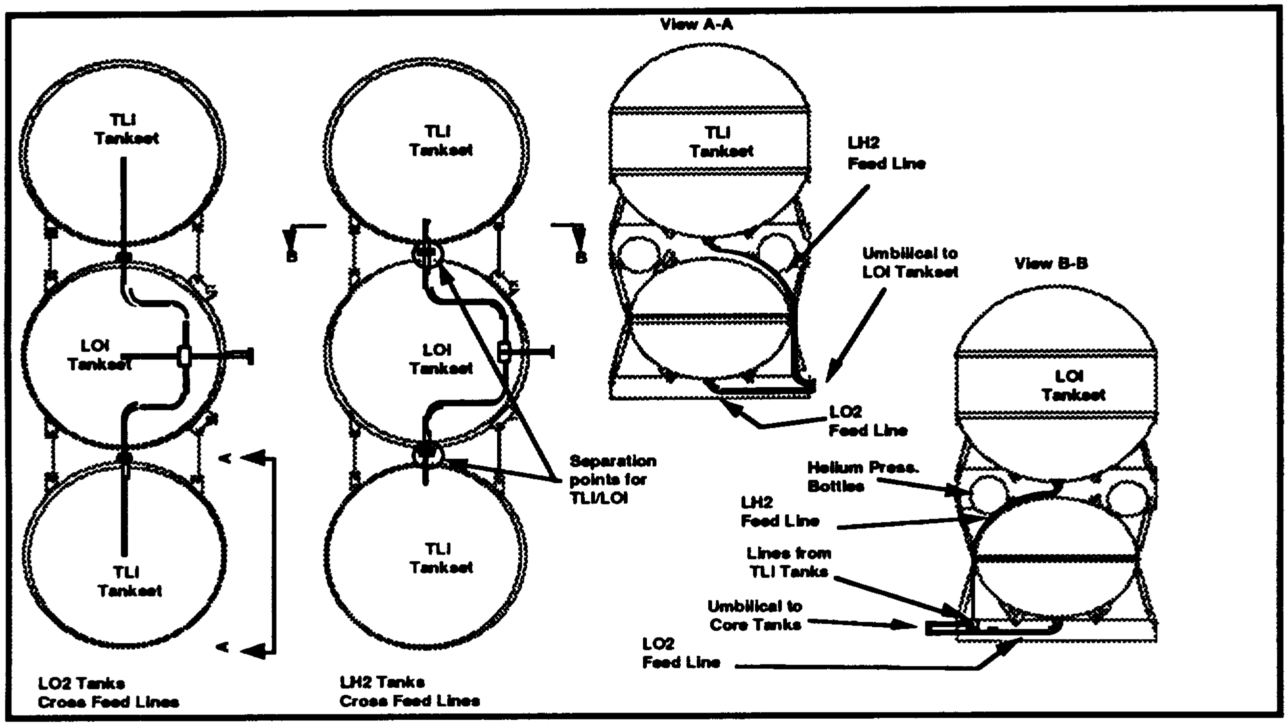


Figure 3.2.2.3-1 Drop Tank Feed Lines

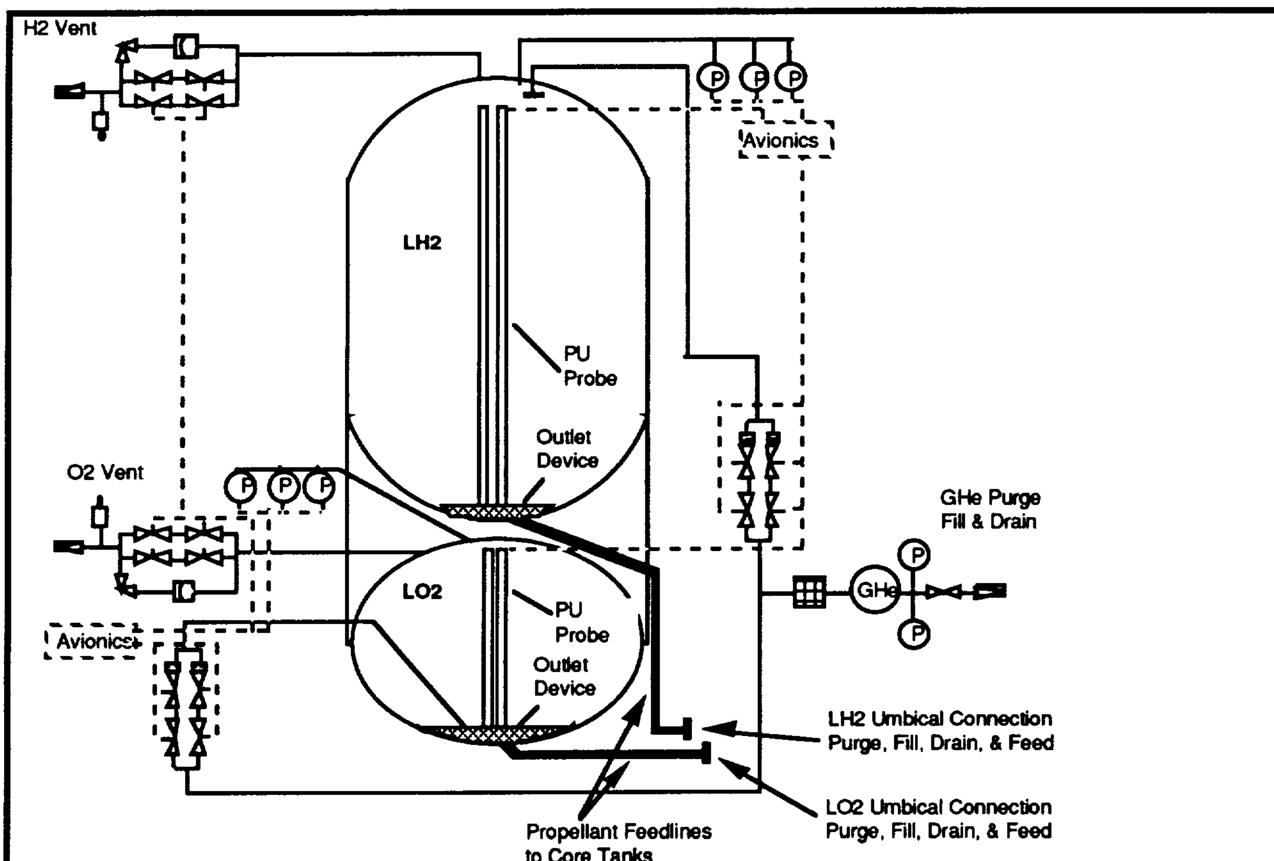


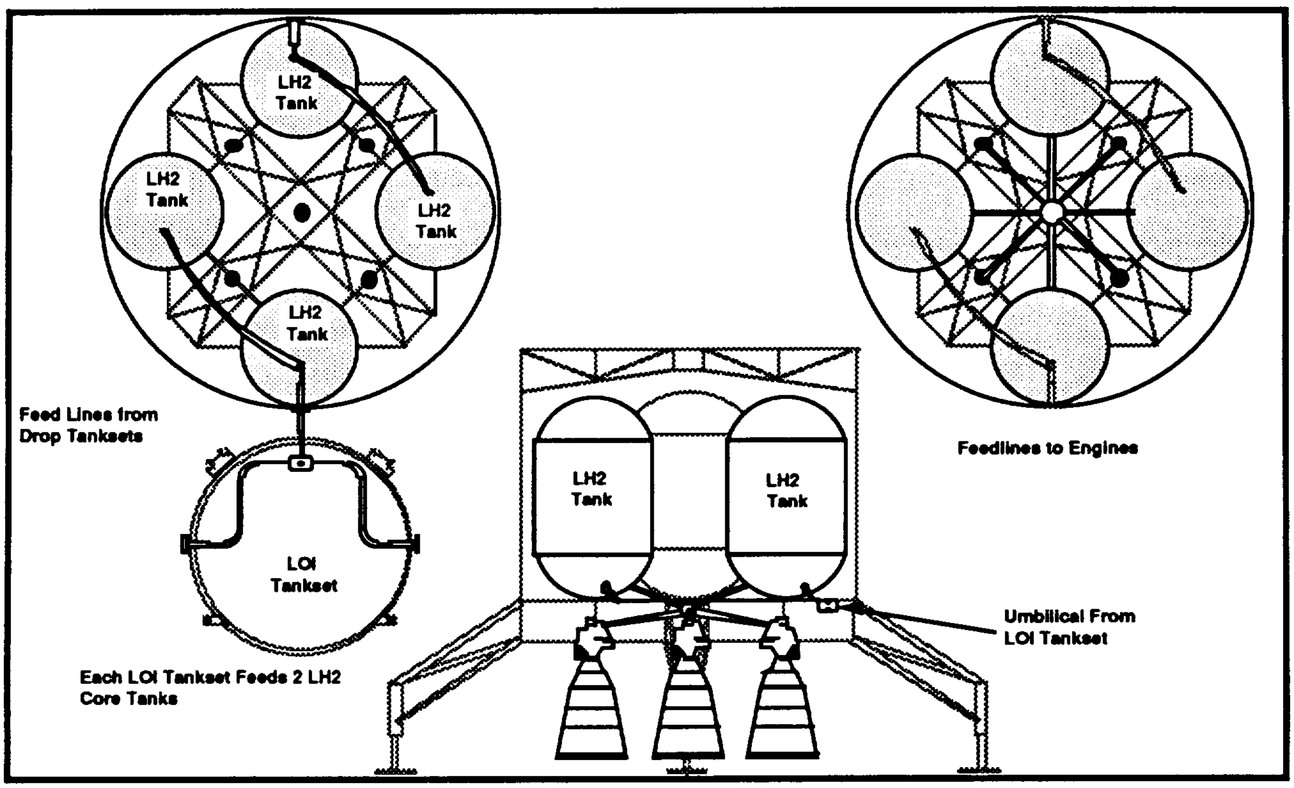
Figure 3.2.2.3-2 Typical Tankset Fluid Schematic

3.2.2.4 Core Tank Feed Lines

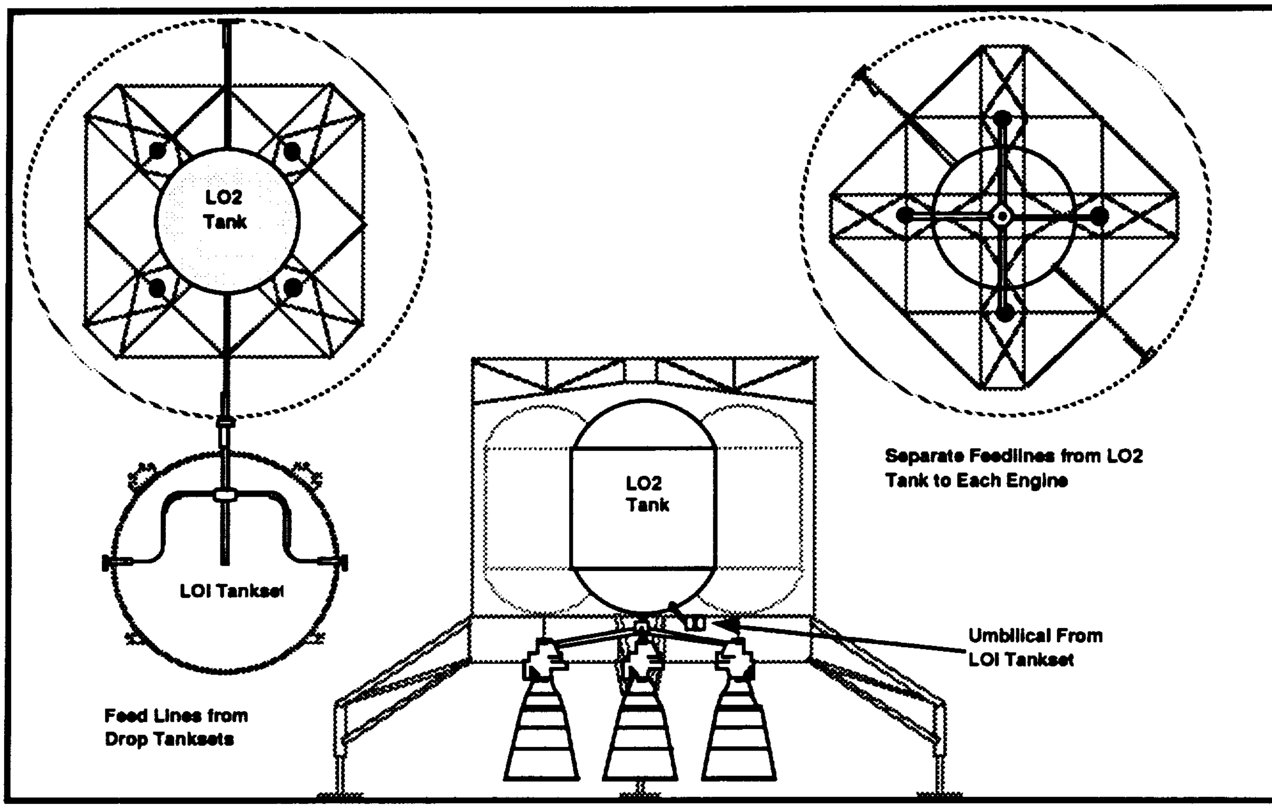
The LH2 and LO2 core tank propellant system schematic is shown in Figures 3.2.2.4-1 and 2.3.2.2.4-2. Propellant is fed from the drop tanksets to the core tanks through the LOI/core tank umbilical, with the two core LH2 tanks fed by one of the LOI tanks. Each LH2 core tank then feeds a manifold with separate feed lines to each individual engine.

3.2.2.5 Return Tank Feed Lines

Figure 3.2.2.5-1 illustrates the flow of propellant from the return tanks in the aerobrake to the core engines. After the core has rendezvoused and docked with the aerobrake, umbilical connections are made at two locations (180° opposite each other) from which separate LO2 and LH2 lines are routed along the core structure



Figures 3.2.2.4-1 Core Tank Propellant System Schematic



Figures 3.2.2.4-2 Core Tank Propellant System Schematic

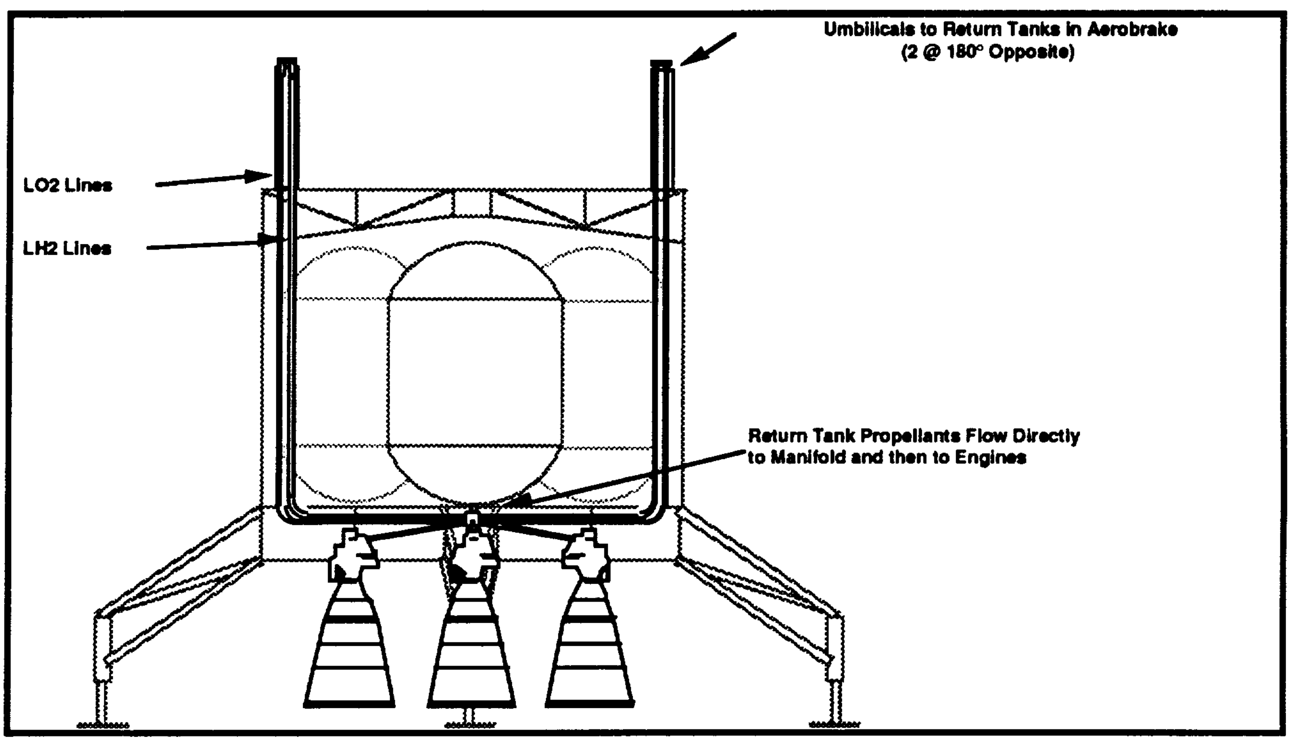


Figure 3.2.2.5-1 Propellant Flow From the Aerobrake Return Tanks

3.2.3 Avionics

The LTS avionics, located in the aerobrake, crew module and the propulsion/avionics core, represents a man rated quad redundant system. The architecture employed assumes that each element operates independently some of the time, requiring each avionics system to stand alone. Details of avionics system functions are discussed in Section 2.3.4.1. The avionics system, located in the propulsion/avionics core, handles all cargo operation functions and interfaces with those elements in the crew module during the piloted operations. This system is composed of two major groups - Guidance, Navigation and Control (GN&C) and Communication and Data Management (C&D Mgmt). Tables 3.2.3-1 & 3.2.3-2 summarize the components, their quantities, and total mass.

Table 3.2.3-1 Guidance, Navigation, & Control

Components	Units	WT	Total
IMU(3 RLG & 3 PMA)	2.00	24.00	48.00
GPS Receiver	2.00	20.00	40.00
GPS Antenna - High	2.00	5.00	10.00
GPS Antenna - Low	1.00	5.00	5.00
EMA Controller	2.00	10.00	20.00
RCS VDA	32.00	0.50	16.00
Guidance & Control Total			139.00
Star Scanner	4.00	6.00	24.00
Navigation Total			24.00
Landing Radar Altimeter	2.00	25.00	50.00
Rendezvous Radar	2.00	25.00	50.00
Landing Radar Electronics	2.00	49.00	98.00
Lander Antenna	2.00	5.00	10.00
Landing & Rendezvous System			208.00
Pan Tilt Cameras	2.00	15.00	30.00
Video Recorders	2.00	15.00	30.00
TV System			60.00
G.N. & C. Core Total			431.00

Table 3.2.3-2 Communication and Data Management

Components	Units	WT	Total
GPS Antenna System	2.00	15.00	30.00
STDN/TDRS Transponder	2.00	15.50	31.00
20W R.F. Power Amp	2.00	6.00	12.00
S-Band R.F. System	2.00	50.00	100.00
UHF Antenna	2.00	10.00	20.00
UHF System	2.00	10.00	20.00
TLM Power Supply	2.00	7.00	14.00
Enclosure Box	1.00	26.55	26.55
Communication			253.55
G.N. & C. Computer	4.00	20.00	80.00
Master Timing Units	2.00	5.00	10.00
Health & Status Computer	4.00	20.00	80.00
TM System	2.00	22.00	44.00
GN &C IU	4.00	10.50	42.00
Enclosure Box	1.00	25.50	25.50
Data Management			281.50
C & D MNGMT Core Total			535.05

3.2.4 Power

Power for the propulsion/avionics core is provided by four fuel cells similar to those aboard STS, but supplied with propellant grade cryogenic reactants. Each fuel cell delivers 12 kw at peak (27.5 V and 436 A) and an average output of 7 kw. 2 kw provides 32.5 V and 61.5 A. The water supplied as a by-product of the fuel cells provides potable water during the mission. Emergency power is provided by Ag-Zn batteries. Table 3.2.4-1 summarizes the power supply components, their quantities, and total mass.

Table 3.2.4-1 Power System - P/A Core

Power System - P/A Core	Qty	Unit Wt lbs	Total
Fuel Cell System	4	86.25	345.00
Radiator System	4	28.75	115.00
Residual H2O System	2	17.25	34.50
Batteries	2	100.00	200.00
Power BUS	4	10.50	42.00
Power Distribution Equipment	4	27.00	108.00
Wiring,Harness, & Connectors	1	100.00	100.00
Enclosure Box	1	15.00	15.00
Total			959.50

3.2.5 Meteoroid & Debris Protection

The meteoroid and debris protection analysis conducted during the STV study determined the best type of protection needed at LEO, the lunar surface, and for the hanger at SSF for the environments STV elements were exposed to. The data in Tables 3.2.5-1 & 3.2.5-2 defines the design requirements used in this analysis.

Table 3.2.5-1 STV - Meteoroid Protection

STV - Meteoroid Protection
• Environment defined in NASA SP 8013 • Direct Impact by Meteoroids - Average Impact Velocity = 20 km/s - Density = 0.5 g/cm^3(loosely packed ice) - Bumper equivalent to 0.015cm aluminum (eg 3 sheets of Beta cloth) plus TPS will protect against meteoroids up to 1 cm in diameter • Lunar Ejecta (from nearby meteoroid impacts) - Average Velocity = 0.1 km/s (1 km/s max) - Average Density = 2.5 g/cm^3 - Flux is hundreds of times larger than for direct meteoroid impacts - Shields equivalent to 0.5 cm thick of aluminum are recommended on core module • Exact velocity distributions should be examined • Composites and ballistic cloths may reduce total shield weight • Low strength/obliquity of ejecta particles may reduce protection requirements

Table 3.2.5-2 STV - Space Debris Protection

STV - Space Debris Protection
• Debris environment estimated for 2004 (average for 2000 to 2010) depends on - Solar Cycle - Altitude (370 km assumed - low is best due to removal of debris by drag) - Growth (5% annually assumed per SSF - 10% considered to be upper limit) - Debris density assumed to be Aluminum • Protection must average over all impact velocities and obliquities - Multiple layer designs will have the total thickness of all layers approximately one-half the diameter of the debris with desired Probability of No Impact (PNI) • Hangar protection needed almost up to limits of debris detection (10 cm) and avoidance - 5 cm debris diameter for 0.9955 probability of no penetration in 10 years (w/ avoidance) - Areal density will be equivalent to thickness of 2.5 cm of aluminum - Total thickness with standoffs approximately 1 meter - Weight = 3 sides $\cdot$ 21m $\cdot$ 21m $\cdot$ 2.5cm $\cdot$ 2.8 g/cm^3 = 92.6 metric tons • Expendable tanks debris protection requirements related to HOURS of exposure per mission - Recommend less than 8 hours exposure per mission (40 hours total for 5 missions) - 1 layer of Beta Cloth plus TPS with 7.5 cm standoff will meet 1 day space debris and 4.5 days meteoroid exposure

Figure 3.2.5-1 shows the flux and particle size differences encountered at each stage of a mission. Since the penetration resistance varies with velocity, density and obliquity, the reliability given by Probability of No Penetration (PNP), has been defined as a reference point to estimate shielding requirements.

Probability of No Impact (PNI) = exp(- Flux x Area x Time) = $e^{-(N \cdot A \cdot T)}$

If "N·A·T" is small (reliability is high), then PNP = 1-N·A·T.

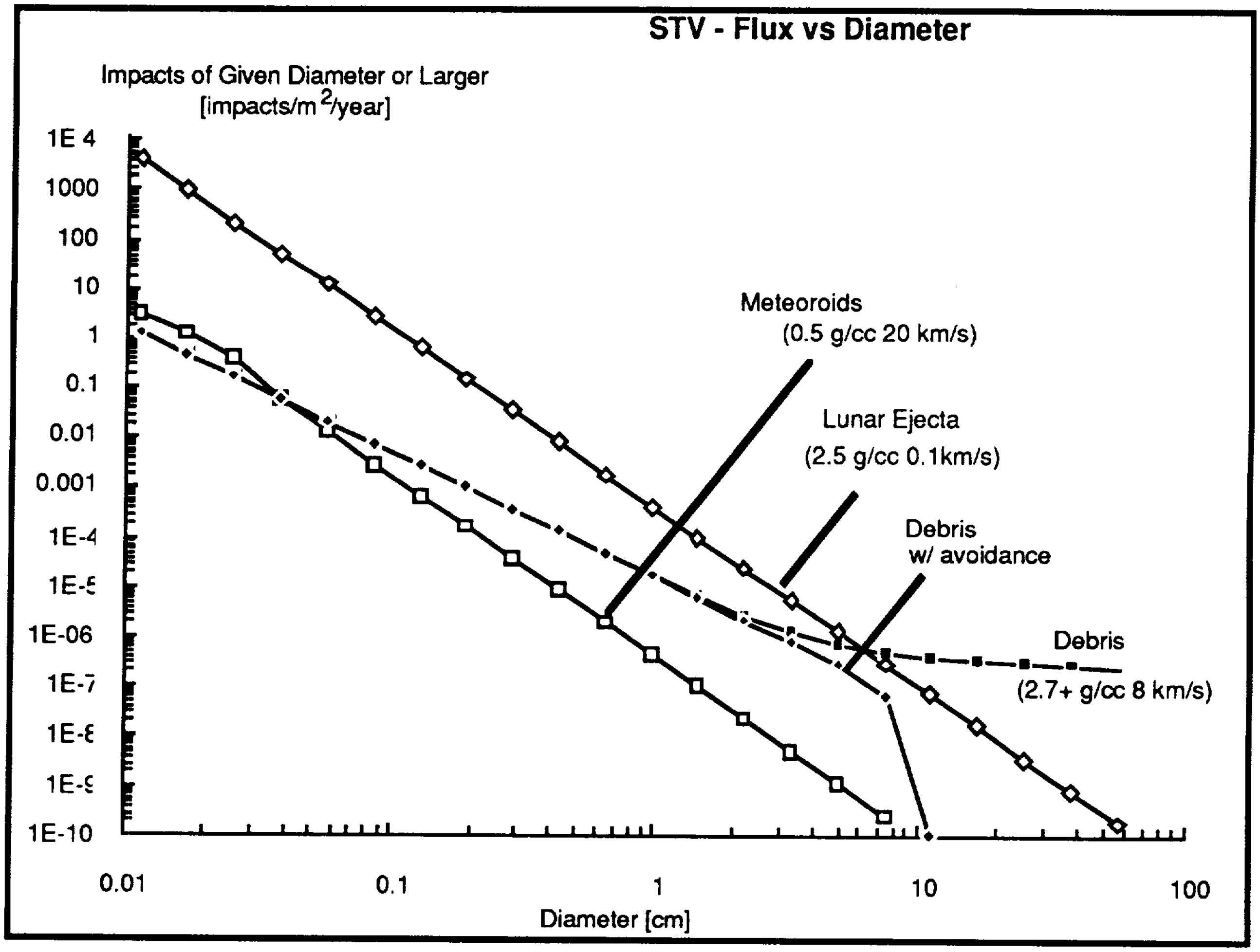


Figure 3.2.5-1 Flux vs Particle Diameter

The overall PNP is obtained by multiplying the PNP from each threat on each element and from each stage of the mission.

Figure 3.2.5-2 defines the particle environment and the critical flux for 0.09955 PNI for key mission phases. The PNP (which covers the entire velocity and obliquity spectrum) for STV elements as well as the threat must be higher than 0.9955 if the overall reliability from impact is to be 0.9955. The shielding recommended for all STV elements accounts for this for preliminary design estimates.

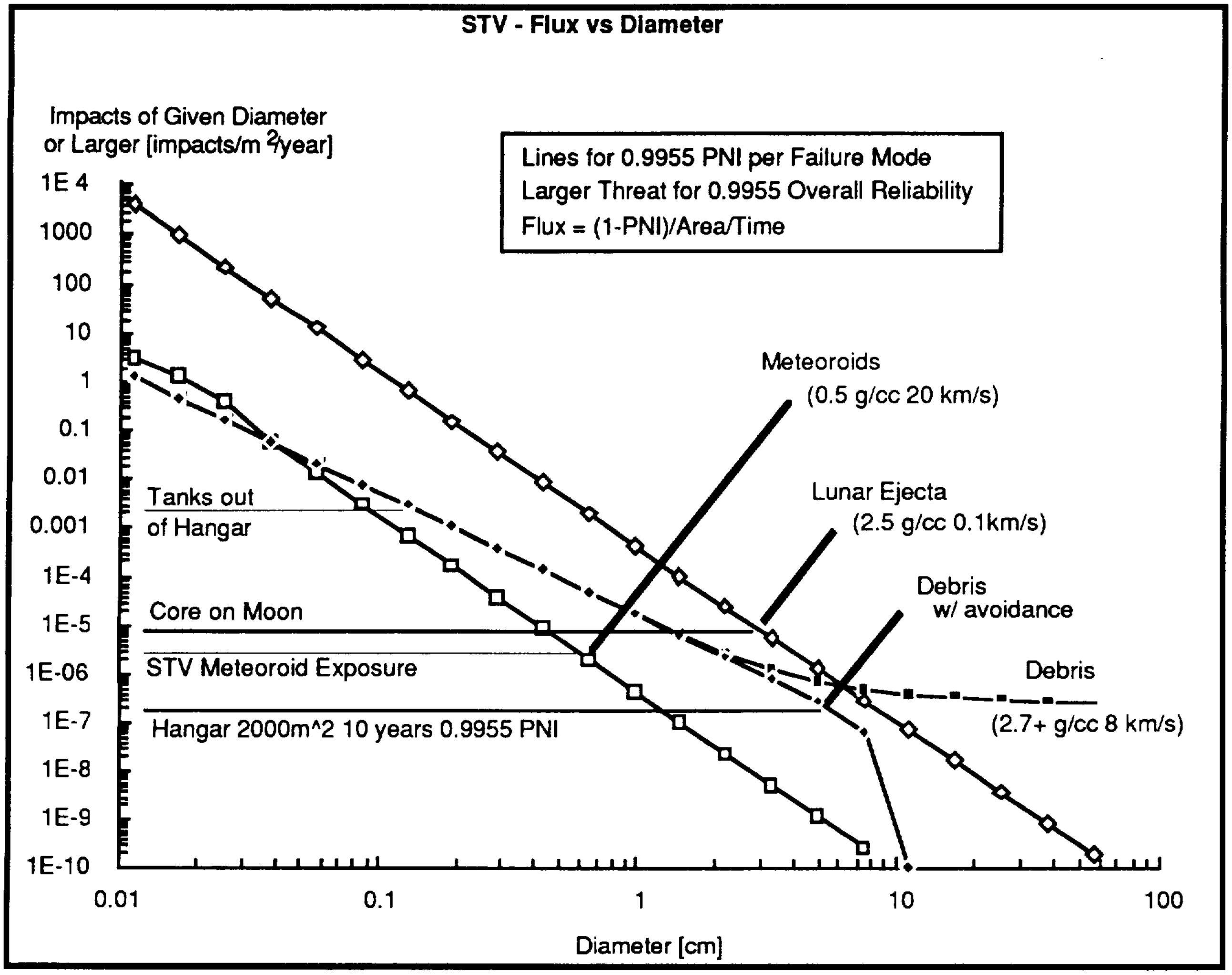


Figure 3.2.5-2 Particle Environment vs Critical Flux

Figure 3.2.5-3 illustrates lunar ejecta vs PNI for the average particle that must be stopped to provide the desired reliability.

$$\text{Probability of No Impact (PNI)} = \exp(-\text{Flux} \times \text{Area} \times \text{Time})$$

$$\text{Total Probability of No Penetration (PNP)} = PNP_{met.} \times PNP_{debris} \times PNP_{lunar\ ejecta}$$

The time estimates used for this analysis are based on a five mission exposure and lunar ejecta that has an average density of 2.5 g/cc and impacts at 0 to 1 km/s. The resulting area estimates include factors for self shielding. Figure 3.2.5-4 illustrates the average particle which must be stopped to provide the desired reliability.

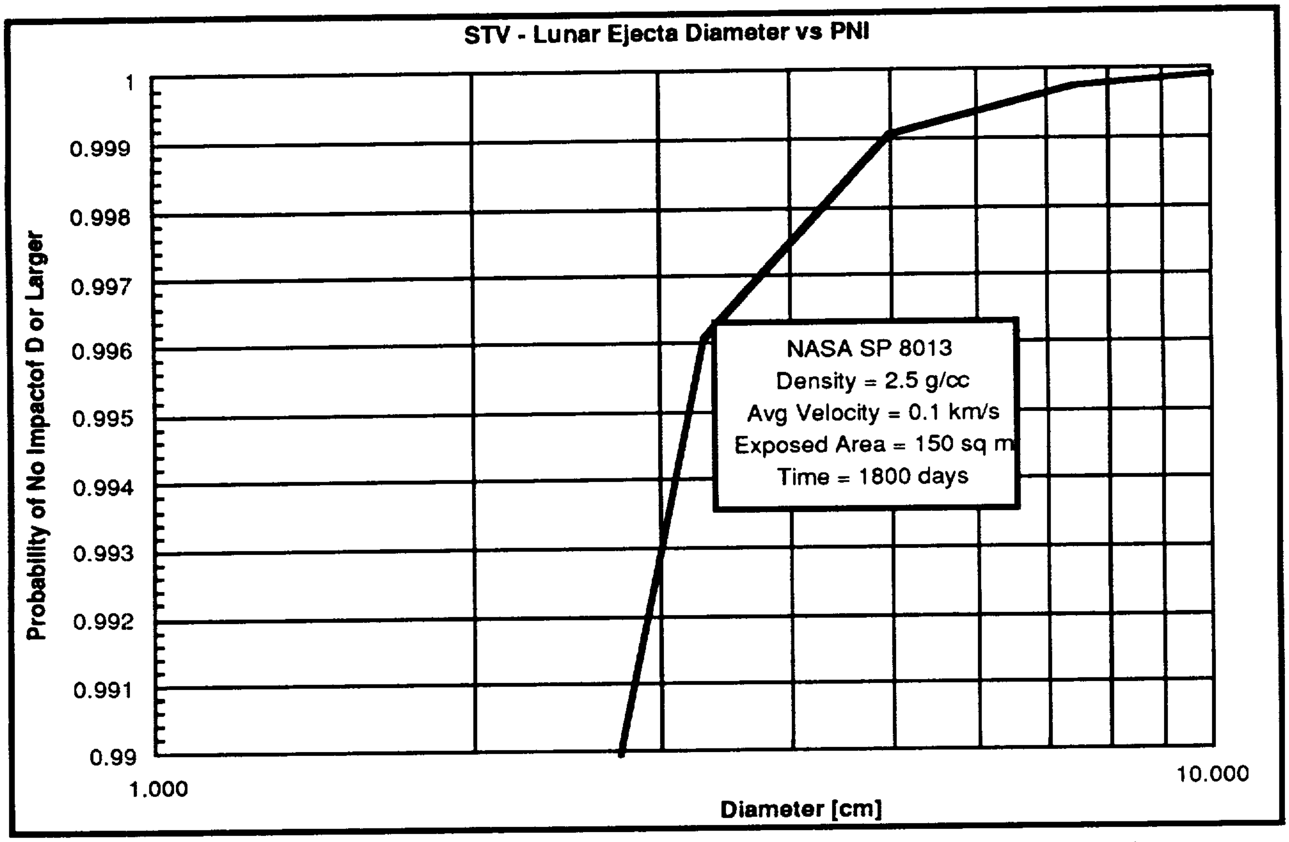


Figure 3.2.5-3 Lunar Ejecta vs PNI for the Average Particle That Must be Stopped

The time estimates used for this analysis are based on a 5 mission exposure, except for 10 years for the hangar, and meteoroids that are predominantly ice particles impacting at 8 to 72 km/s. The area estimates include factors for self shielding and view factors due to shielding by the earth or moon.

Figure 3.2.5-5 illustrates the average space debris particles which must be stopped to provide the desired reliability.

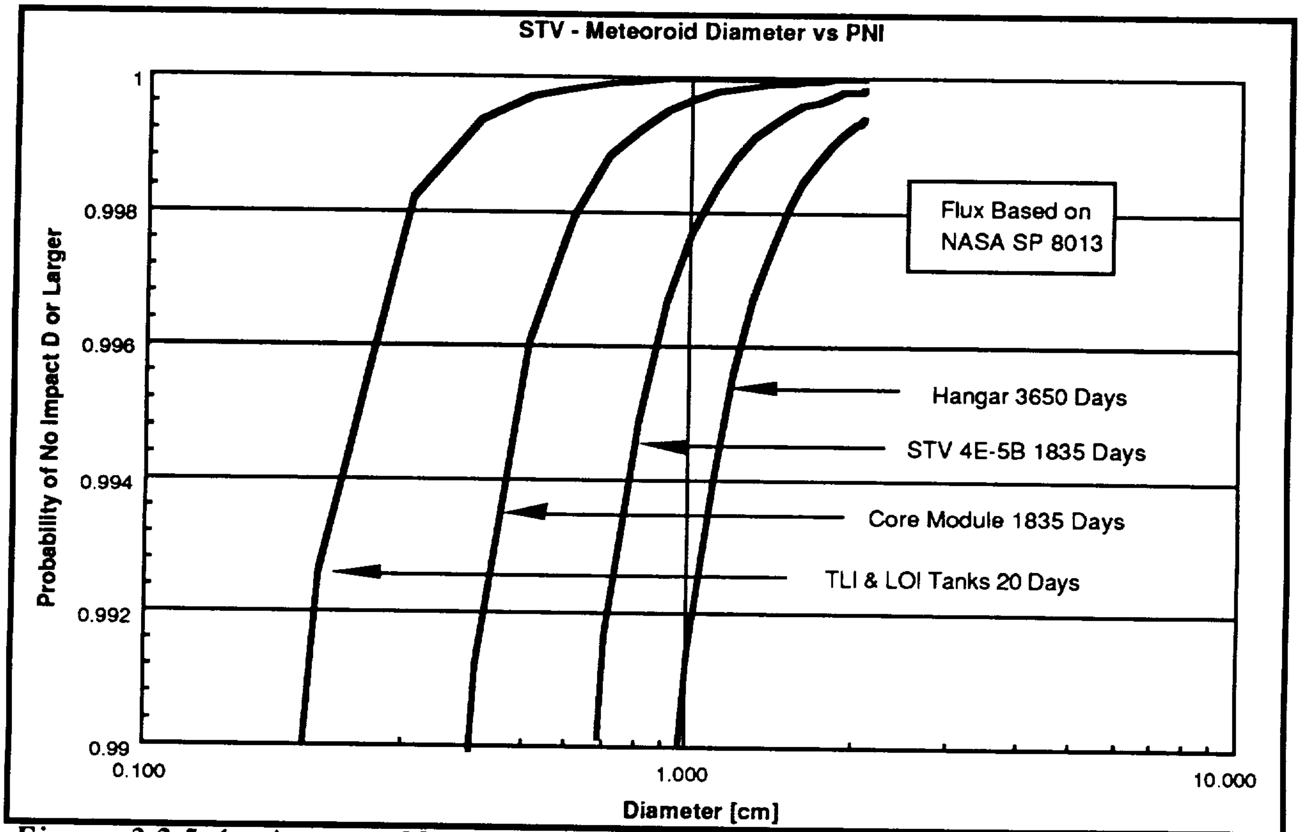


Figure 3.2.5-4 Average Meteoroid Particles Which Must be Stopped

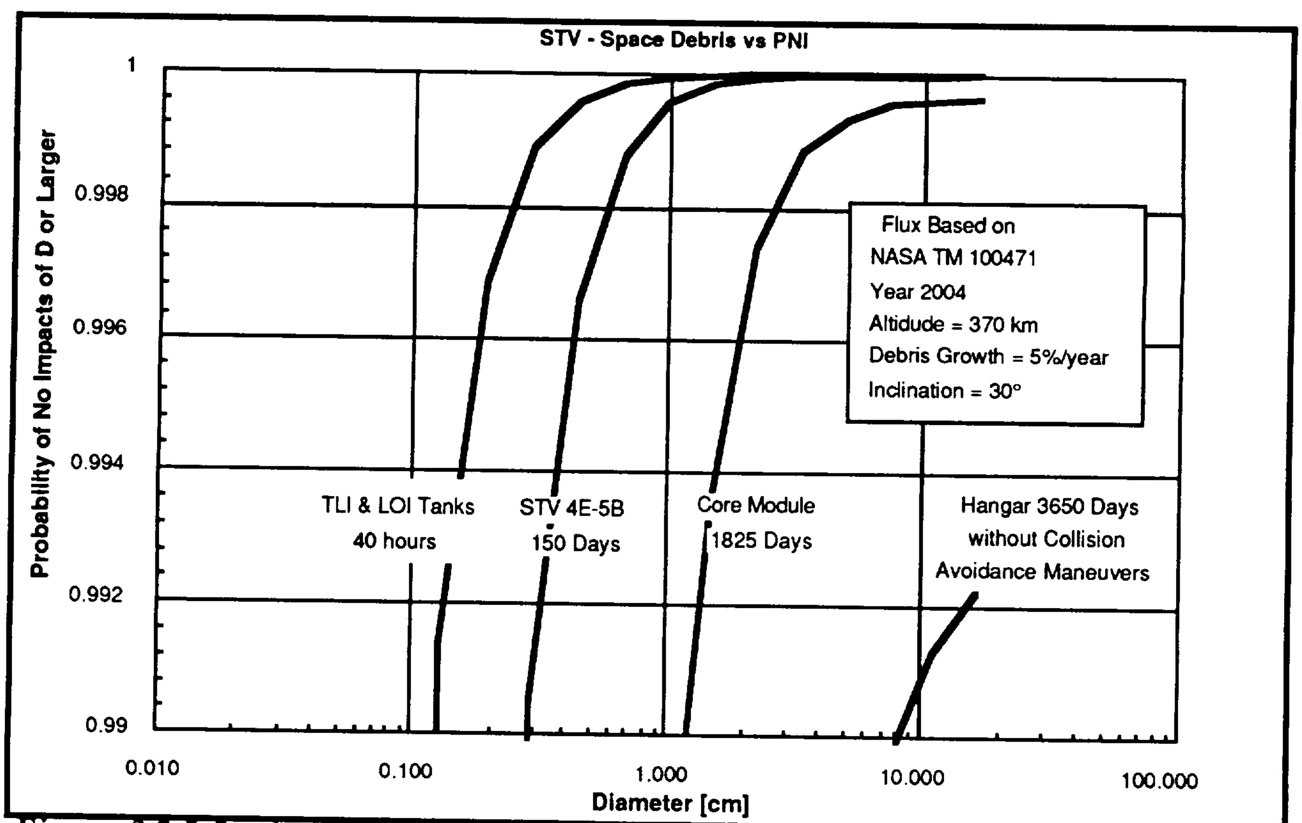


Figure 3.2.5-5 Average Space Debris Particles Which Must be Stopped

The time estimates used for this analysis are based on a 5 mission exposure, except for 10 years for the hangar, and space debris that is predominantly aluminum particles impacting at 0 to 16 km/s in LEO. The area estimates include factors for self shielding. Debris threat depends on the growth rate with time as well as the altitude. Threat is reduced at lower altitudes due to the removal of debris particles by atmospheric drag.

Shield requirements address the entire threat spectrum including particle size, impact velocities and obliquity versus the performance of optimized multilayer shield designs. Table 3.2.5-3 provides a method of estimating shield thickness and spacing as a function of the particle size estimated from Figures 3.2.5-1 through 3.2.5-5. Multi-wall shields are not as effective at 3 km/s or for 45° obliquity impacts as they are for normal impacts at 7 km/s since the debris particle does not fragment as well, therefore the total weight of the shield increases to account for the non-optimum performance. The design of the hangar shield uses multi-wall designs developed under Martin Marietta IR&D, and under contracts from NASA and the U.S. Air Force Defensive Shields Program. The lunar ejecta shield thickness estimate is very preliminary at this time with additional data to be provide as it becomes available. Composites or ballistic cloth may be much more effective in stopping that velocity of particle than the estimated weight of monolithic aluminum.

3.3 Aerobrake

The aerobrake provide the thermal protection for the LTS during the aeropass maneuver before returning to SSF. Studies have determined that the aerobrake design provides a sizable savings in propellant, directly translate into a cost savings. Another studies analyzed different type of aerobrake construction and recommended a rigid, hard shell aerobrake design. Further analysis of on-orbit assembly of a rigid hard shell aerobrake found that fewer pieces requiring assembly was desirable which resulted in the three piece folding concept. As part of this study, the manifesting of the folding aerobrake in the ETO launch vehicle was considered and found to be compatible with a 7.6 m payload envelope. An isometric view of this rigid aerobrake structure is shown in Figure 3.3-1(20).

Operation of the lunar mission requires the aerobrake and the lander to separate in LLO before the lander makes the lunar descent, leaving the aerobrake in a 60 x 100 nm orbit. This requires that the aerobrake have station keeping, rendezvous, and docking capabilities. This is accomplished by converting the aerobrake from a passive element to an active vehicle using its own avionics,

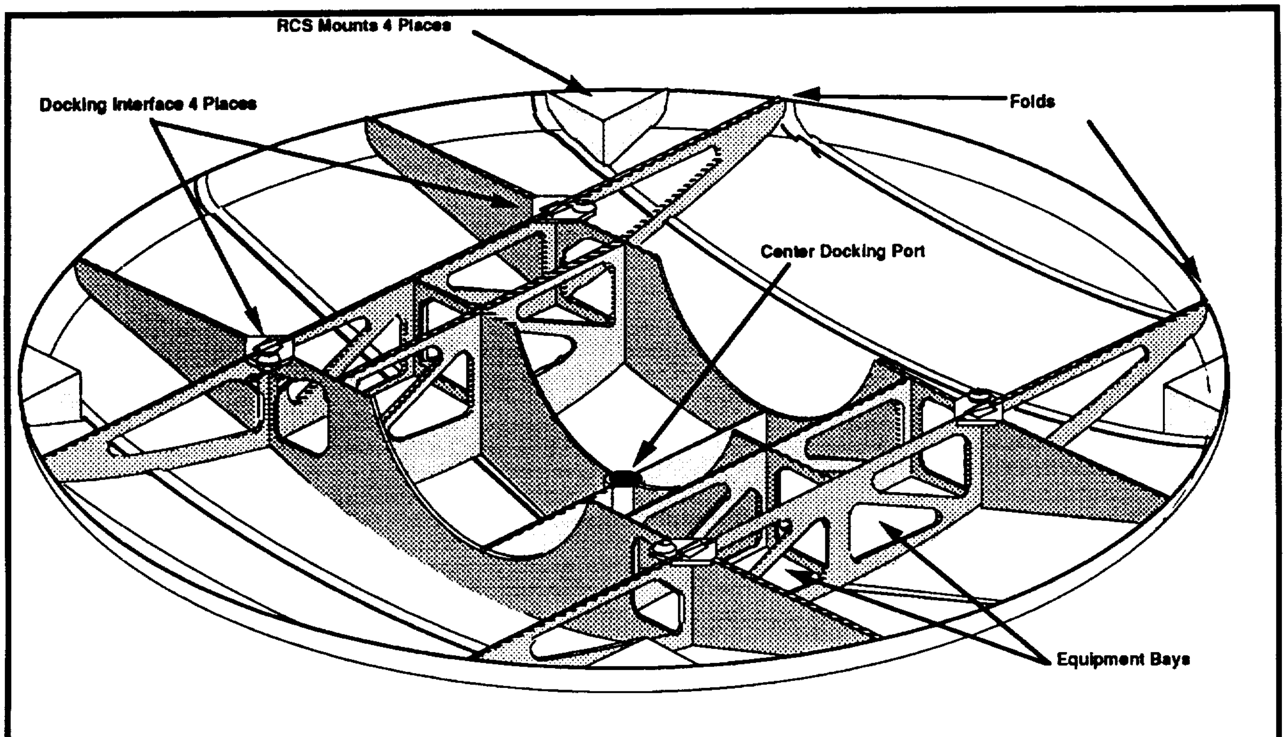
Table 3.2.5-3 Shielding Requirements As a Function of Particle Size

STV - Shield Requirements			
• Areal Density of Shield is Proportional to Diameter of Impacting Particle			
	Equivalent Total Thickness of Aluminum	Areal Density kg/m^2 (D in cm)	MinimumBumper Standoff
Space Debris	0.75 D	20D	20 D
Meteoroids	0.15 D	4 D	10 D
Lunar Ejecta	0.15 D	4 D	Not Sensitive
• Total Shield Thickness and Density includes TPS and Rear Wall			
• Optimum Designs may Require Multiple Layers or Geometric Disruptors (developed on IRAD, NASA, and Air Force/Defensive Shields Programs)			
• Debris Shield Thickness Accounts for Reduced Resistance to Oblique (45°) and High Velocities (16 km/s) or Low Velocities (3 km/s)			

power, and RCS for control. The following sections detail the structural elements and the subsystems associated with the aerobrake.

3.3.1 Structure

The aerobrake is a graphite-polyimide structure with overall dimensions of 13.72 m in diameter and 2.59 m in depth, covered with shuttle type ceramic tiles (FRICS-20). The structural details are shown in Figures 3.3.1-1, 3.3.1-2, and 3.3.1-3. Two major longitudinal and three major transverse bulkheads provide the primary structural elements, with additional frames and intermediate bulkheads for support. The bulkheads are fabricated from graphite-polyimide face sheets and a foam core and the frames are extruded graphite epoxy "T"-sections. The surface panels are formed from graphite-polyimide face sheets with an aluminum honeycomb core. The center section panels are 0.51 cm thick and the outer panels are 0.38 cm thick and are mounted to the surface panels extruded graphite epoxy angles.



13.72 m (45 ft) Diameter Rigid Aerobrake
Folds in 2 places

Figure 3.3.-1 Rigid Aerobrake Isometric

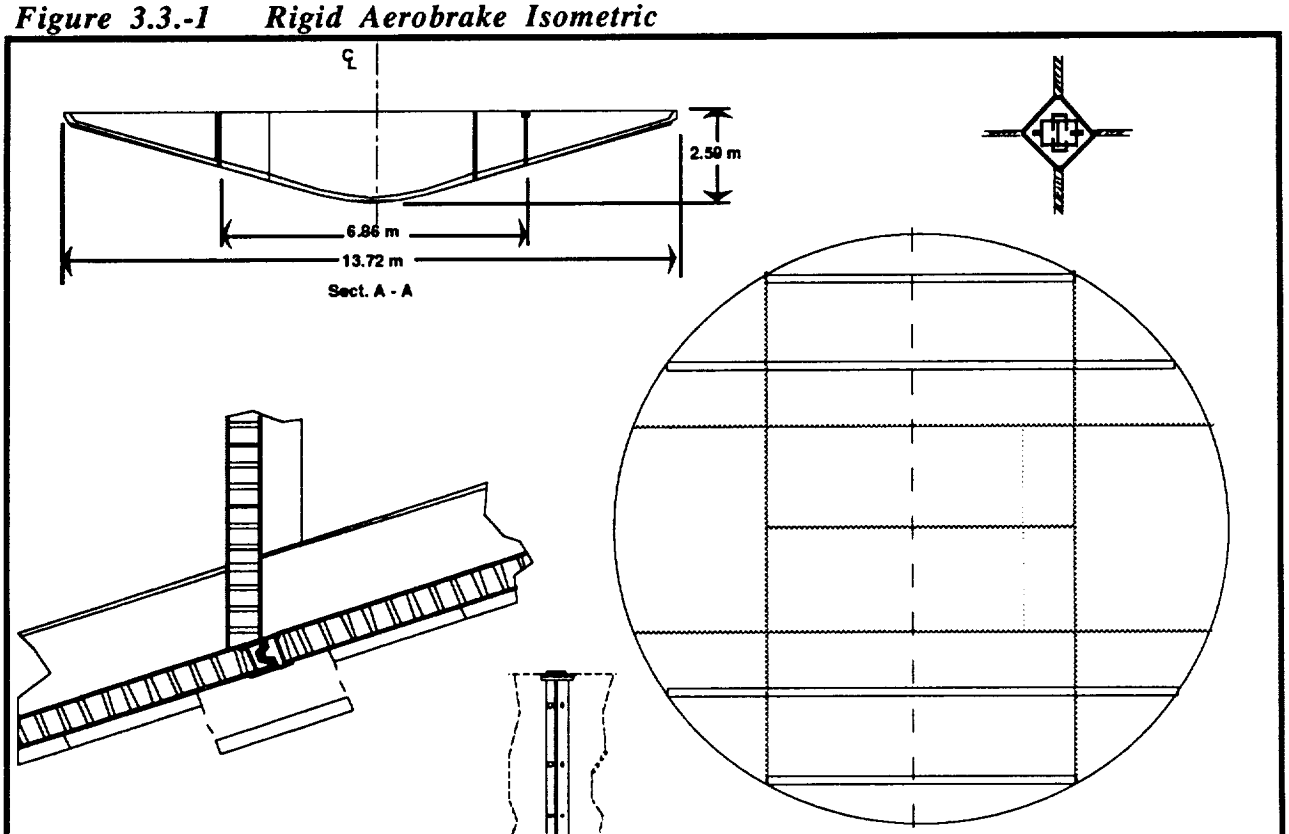


Figure 3.3.1-1 Aerobrake Structure Details

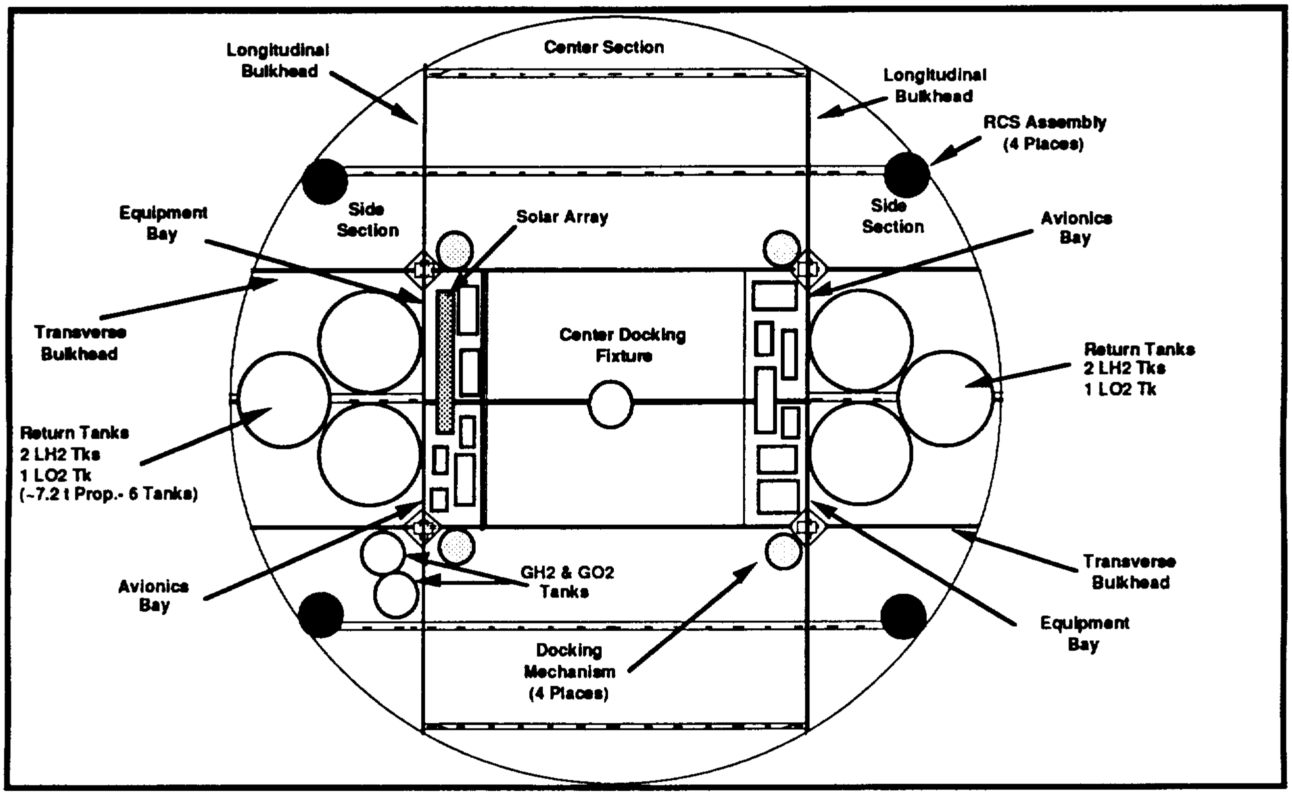


Figure 3.3.1-2 Aerobrake Structure Details

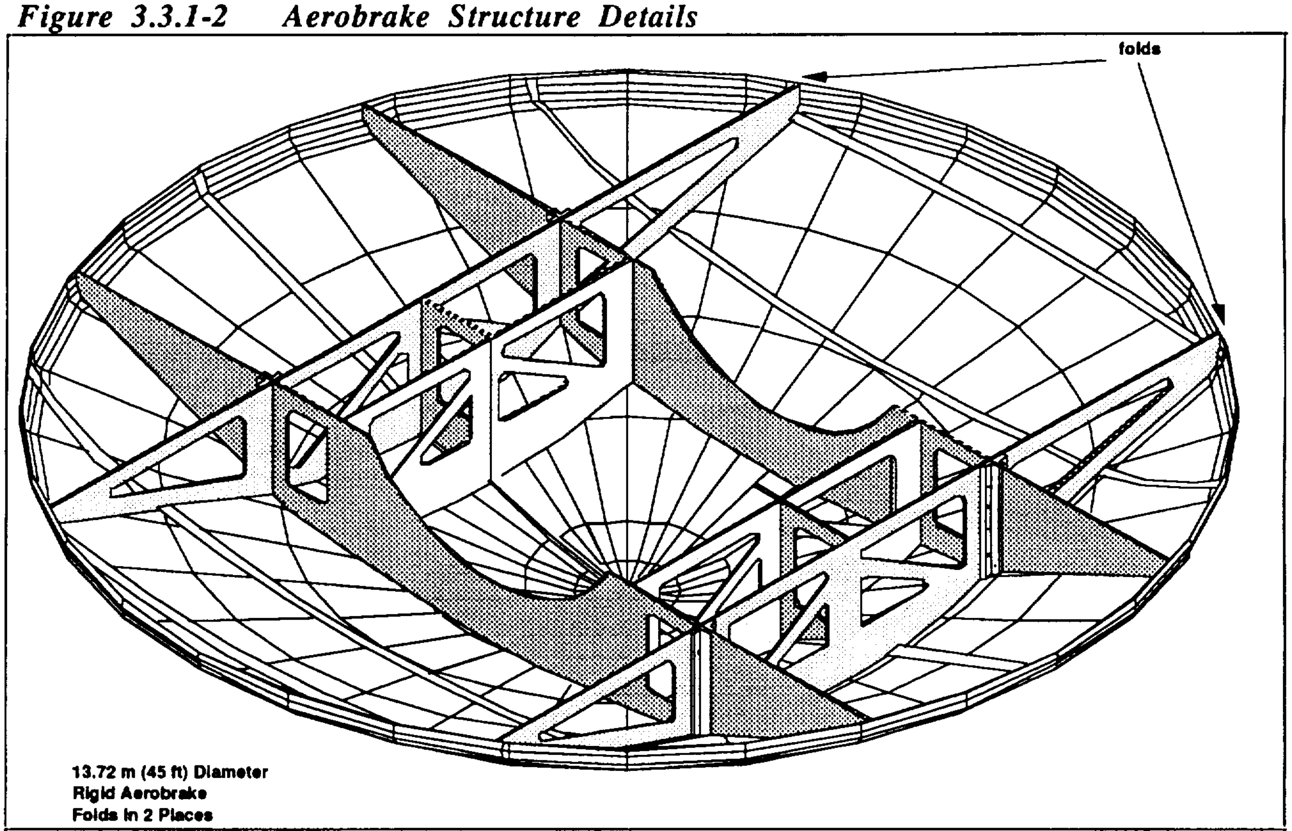


Figure 3.3.1-3 Aerobrake Structure Details

LEO assembly of the aerobrake is performed by rotating the two outer sections into place about hinges located at the intersection of the longitudinal and outer transverse bulkheads. Proper alignment to the center section is assured by a male/female aluminum joint along the intersecting surface panels. The outer section is then secured into place through the use of locking pins located on the outboard side of the longitudinal bulkheads. A section of the outer ceramic tile around the interface area is initially not installed to allow the hinged motion required for deployment. Once the side sections are deployed, the ceramic tile will be installed on orbit over the interface area.

3.3.3 Subsystems

The aerobrake is left in a 60 to 100 nm orbit when the lander separates for descent to the lunar surface. In order for the aerobrake to maintain its position and be able to rendezvous and dock with the lander for the return trip, it had to be outfitted with the necessary components to perform this part of the mission. The location of the equipment contained within the aerobrake structure is shown in Figure 3.3.3-1(24). Avionics bays and equipment bays are located along either side of the longitudinal bulkhead. The docking equipment is located on the central bulkhead and at the intersection of outer transverse bulkheads and the intermediate longitudinal bulkheads. The following section deals with the subsystems located on the aerobrake.

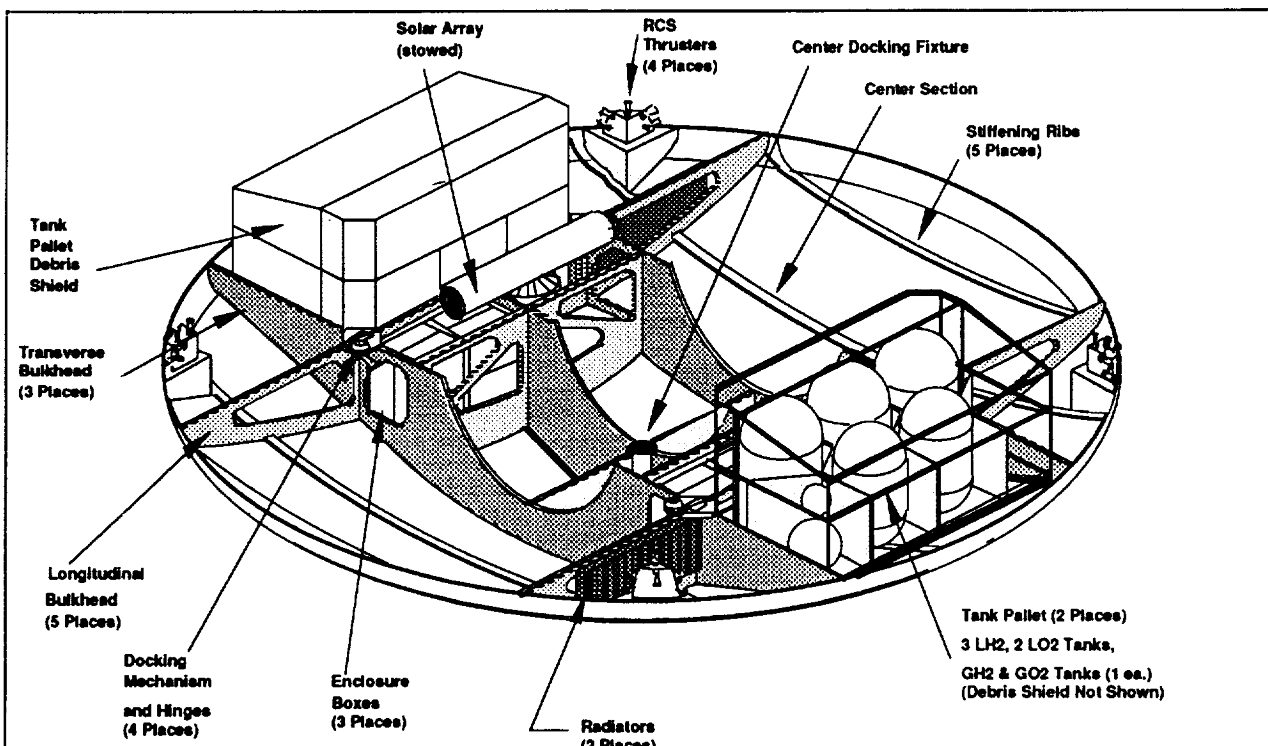


Figure 3.3.3-1 Avionics/Aerobrake Equipment Relationship

The aerobrake also houses the return propellant for the lander. This is located in two tank pallets consisting of 3 LH2 tanks and 2 LO2 tank in each pallet. The pallets are positioned in the outer sections of the aerobrake leaving the center section free for mating the lander and crew module to the aerobrake.

3.3.3.1 Avionics

The aerobrake avionics subsystem must meet the same design requirements (quad redundant man rated) as found in the core vehicle. The function of the system is to provide the aerobrake with station keeping and rendezvous and docking capabilities. The operational details of the subsystem are discussed in the avionics trade study section in this report. Tables 3.3.3.1-1 and 3.3.3.1-2 list the components of the GN&C and C&D handling system respectively.

Table 3.3.3.1-1 GN&C System Aerobrake

G N & C System Aerobrake	Qty	Unit Wt lbs	Total
IMU	2	24.00	48.00
Rendezvous Radar	2	25.00	50.00
Star Scanner 2	2	6.00	12.00
RCS VDA	32	0.50	16.00
Total			126.00

Table 3.3.3.1-2 C & D Handling System Aerobrake

C & D Hndlg System Aerobrake	Qty	Unit Wt lbs	Total
G.N. & C. Computer	2	20.00	40.00
Health & Status Processors	2	20.00	40.00
TDRS Transponder	2	15.50	31.00
20W R.F. Power Amp	2	6.00	12.00
S-Band R.F. System	2	50.00	100.00
TLM Power Supply	2	7.00	14.00
TM System	2	22.00	44.00
GN &C IU	2	10.50	21.00
Enclosure Box	1	35.55	35.55
Total			337.55

3.3.3.2 Power

The power requirement for the aerobrake was estimated to be 1.75 kw while in LLO. Fuel cells were first considered, however thermal control and disposal of the fuel cell by-product, water, posed too many problems. A combination of batteries and solar array (Figure 3.3.3.2-1) for the power supply was then proposed. A flexible substrata solar array design with a surface area of ~210 sq ft was chosen. For stowage, the array panels are hinged together to fold into a stack like an accordion. A motor-driven lightweight coilable mast assembly automatically unfolds the panels, tensions the array to hold it flat and retracts and refolds the panels before mating with the lander.

Ag-Zn batteries were chosen for the backup power supply when the array is not receiving solar light or is retracted. Additional equipment such as power control unit, current charger, and power distribution unit complete the power subsystem for the aerobrake. The electrical power equipment breakdown is given in Table 3.3.3.2-1.

3.3.3.3 Return Tanks

The return propellant for the lander is stowed in the aerobrake while the lander is on the lunar surface. This approach was taken to avoid the penalty of having to expend extra propellant to carry the mass of the return propellant to the lunar surface and then lift it off again. The propellant (7.1 mt) is divided between two sets of tanks housed in a pallet mounted on the outboard port and starboard sections of the aerobrake. Figure 3.3.3-1 shows the arrangement of the tanks in each of the pallets and the structural design of the pallets. The structure is composed of graphite epoxy struts and graphite polyimide honeycomb bulkheads.

A debris shield of graphite polyimide honeycomb panels with SOFI covers the propellant tanks. These pallets are mounted into place on the aerobrakes using trunnion and locking devices similar to the STS arrangement.

The return tankset consists of three LH2 tanks and two LO2 tanks. The tanks are fabricated from aluminum lithium spun domes and isogrid barrel panels. MLI surrounds the tanks for thermal control. A set of accumulator tanks (GO2 and GH2) are also mounted in the pallet to collect the

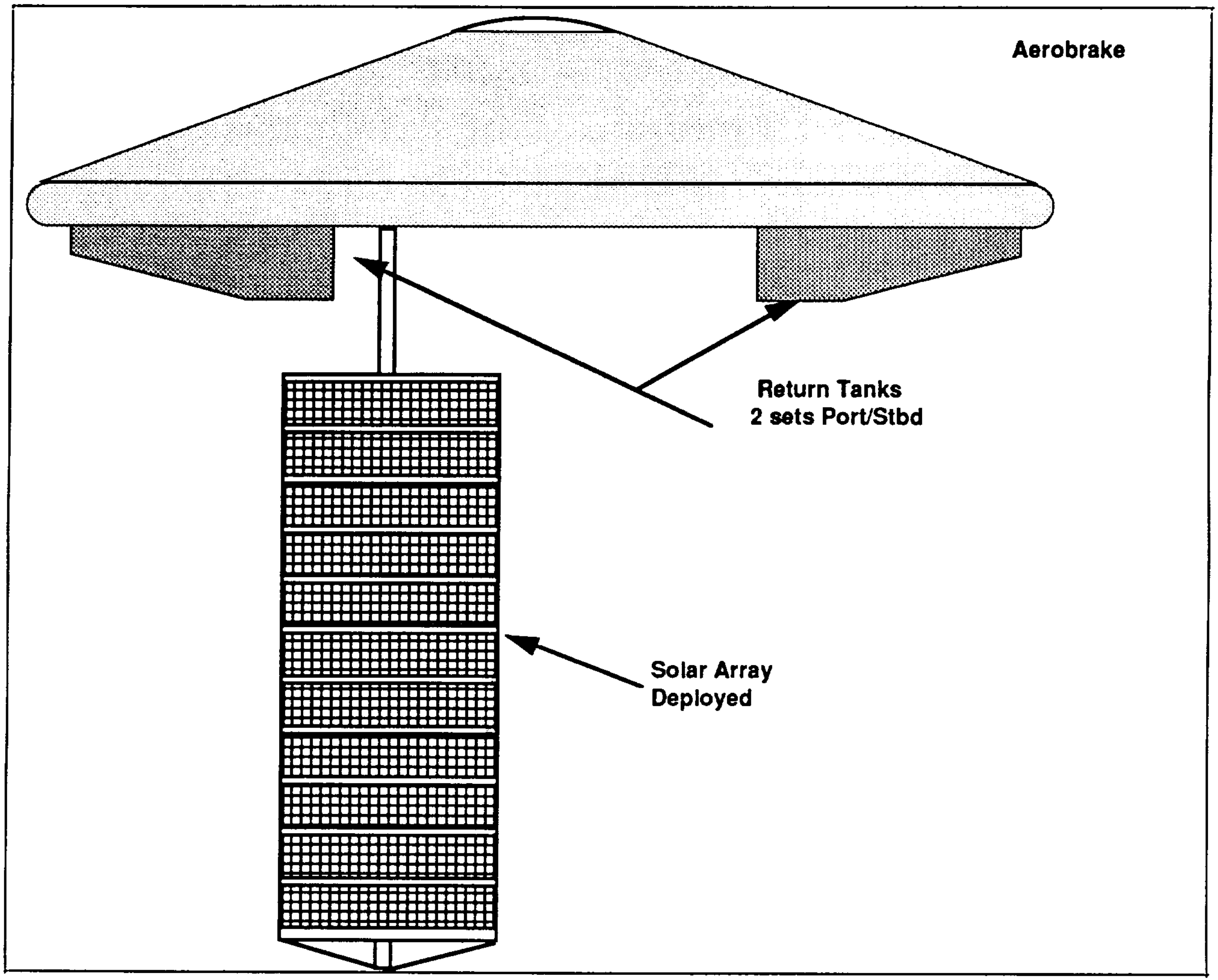


Figure 3.3.3.2-1 Aerobrake Batteries and Solar Array

Table 3.3.3.2-1 Power System - Aerobrake

Power System - Aerobrake	Qty	Unit Wt lbs	Total
Solar Array	1	200.00	200.00
Deployment Control Electronics	2	19.00	38.00
Drive Electronics	2	12.00	24.00
Power Distribution Box	2	25.00	50.00
Power Control Unit	1	120.00	120.00
Current Charger	2	3.00	6.00
Batteries	2	90.00	180.00
Wiring,Harness, & Connectors	1	83.30	83.30
Enclosure Box	2	20.90	41.80
Radiator System	2	28.75	57.50
Total			800.60

boiloff from the propellant tanks for use with the RCS thrusters. All the necessary fill, feed, and vent lines are included in the tank pallet. Quick disconnect umbilical connections allow for easy mating to the propellant lines in the aerobrake and docking with the lander.

3.3.3.4 RCS

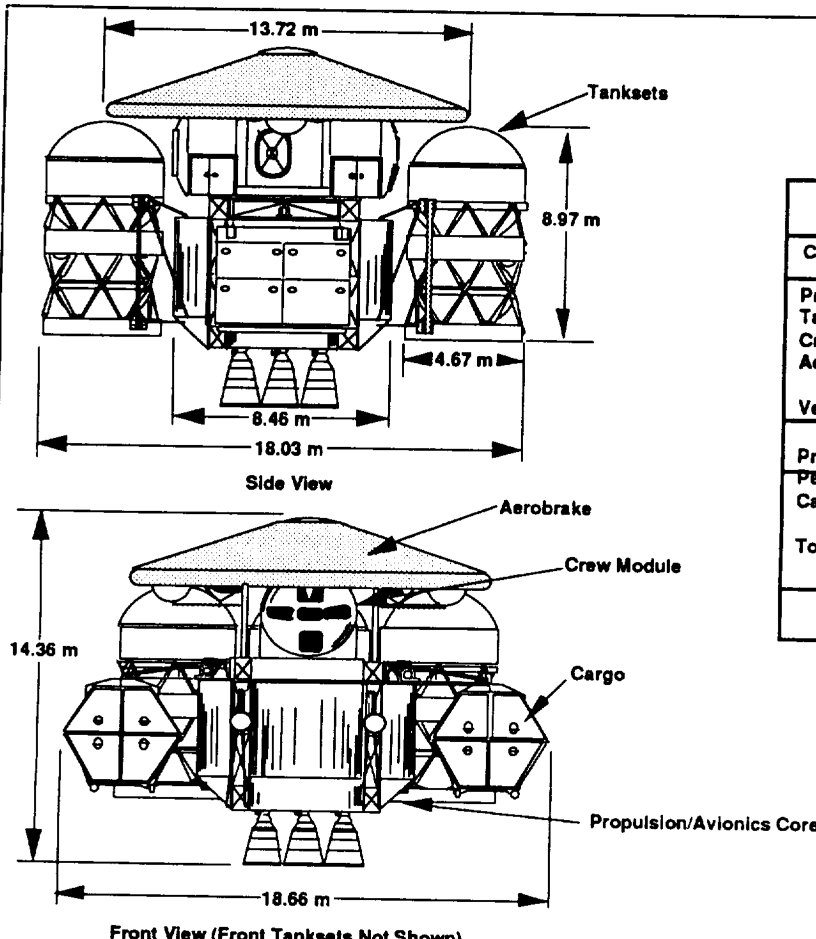
The location of the reaction control system thrusters on the aerobrake is shown. Four clusters (six RCS thrusters each) are placed along the outer edge of the aerobrake to assist in performing aeroassist, docking, and attitude control maneuvers. Variable throttle thrusters are desirable to eliminate the need for multiple systems. Care must also be taken to ensure the thrusters are located within the wake of the aerobrake for the return trip to station.

3.3.4 Mass Properties

Table 3.3.4-1 gives the top level breakdown of the aerobrake structure and subsystem components. The components weights are given in pounds, kilograms and metric tons, (LBS, KG, and MT).

Table 3.3.4-1 Aerobrake Summary

AEROBRAKE SUMMARY DESCRIPTION	WEIGHT LBS	MASS KG	MASS M.T.
STRUCTURE	2336	1059	1.06
TPS	1512	686	0.69
MECHANISM	714	324	0.32
PROPELLANT TANKS	506	230	0.23
RCS SYSTEM	270	122	0.12
G. N. & C.	126	57	0.06
COMMUNICATION & DATA HNDLG	391	177	0.18
ELECTRICAL POWER	778	353	0.35
THERMAL CONTROL SYSTEM	74	33	0.03
GROWTH (15%)	1006	456	0.46
DRY WEIGHT	7713	3498	3.50



Mass Properties	
Components	Mass (t)
Prop/Avionics Core	7.1
Tanksets (4 TLI & 2 LOI)	9.1
Crew Module	7.7
Aerobrake & Equip	3.5
Vehicle Dry Mass	27.5
Propellant	174.
Personnel/Misc	.6
Cargo w/Sppt	15.2
Total Mass	217.5

Figure 3.4-1 STV Piloted Configuration Dimensional Detail

3.4 Piloted Configuration

This section deals with those components unique to the piloted configuration and the some of the mission operations. The STV piloted configuration is designed to carry a crew of four and 14.6 mt of cargo using 174 mt of propellant between the various tanks. The vehicle's overall and dimensions are 14.36 m by 18.66 m by 18.03 m (Fig. 3.4-1) when fully assembled and ready to leave from LEO. The piloted vehicle consist of a crew module, cargo modules and support structure, the two drop tanksets (three tanks per side), and an aerobrake with its associated equipment mounted to the propulsion/avionics core module.

3.4.1 Mass Properties

Table 3.4.1 gives the top level mass properties breakdown for the piloted vehicle at ignition as the vehicle is ready to leave from LEO.

Table 3.4.1-1 Mass Properties Breakdown Piloted Configuration

	DESCRIPTION	MASS	MASS	MASS
	PILOTED SUMMARY AT IGNITION	KG	KG	M.TONS
02	STRUCTURE		2363.15	2.36
03	PROPELLANT TANKS		8951.84	8.95
04	PROPULSION SYSTEM		380.34	0.38
05	MAIN ENGINES		1150.11	1.15
06	RCS SYSTEM		122.45	0.12
07	G. N. & C.		252.61	0.25
08	COMMUNICATION & DATA HNDLG		395.78	0.40
09	ELECTRICAL POWER		796.87	0.80
10	THERMAL CONTROL SYSTEM		891.67	0.89
11	AEROBRAKE		2068.91	2.07
19	GROWTH		2606.06	2.61
	DRY WEIGHT		19979.79	19.98
12	CREW MODULE		7789.17	7.79
15	PROPELLANTS		174139.68	174.14
01	CORE TANKS	31700.00		
02	DROP TANKS TLI	112900.00		
03	DROP TANKS LOI	22113.61		
04	RETURN TANKS	7300.00		
05	FLUIDS & PRESSURANTS	126.08		
17	PERSONNEL		655.00	0.66
18	CARGO		15144.22	15.14
	TOTAL WEIGHT		217707.86	217.71

3.4.2 Crew Module

The crew module is required to support a crew of four during the five to six day trans-lunar and trans-Earth flight and support the crew for the first 48 hours on the lunar surface. Some of the general structural and accommodations requirements for the crew module are:

a) Designed for 5 g loading
b) Two hatches to be provided
c) Capable of berthing to SSF
d) Must fit within the aerobrake wake
e) Meteoroid shield to be used
f) Checkout, repair, and resupply is done at SSF
g) ALSPE shelter to be provided
h) Allow for 2 repressurizations

i) At least 6 cubic meters per person of habitable volume
j) Stored oxygen with regenerable molecular-sieve bed CO2 removal
k) 14.7 psi for normal operations
l) 1.8 kg of food and 2.0 kg of water per man per day
m) Avionics and power interfaces with core module

The general description of the crew module (Figure 3.4.2-1 30) is approximately 72 cubic meters in volume and 8.54 m long by 3.67 m in diameter. The crew module is mounted to the propulsion/avionics core with trunnions and keel fittings similar to those used on the STS system. The module is divided into three major sections - the forward section which houses the flight deck, the mid section which serves as EMU storage, storm shelter, and lunar egress, and the aft section which houses the waste management system, the food preparation system, and station berthing. The crew module can also be utilized at SSF as an additional work station and can be utilized on the lunar surface as a remote habitat and/or safe haven. Unpressurized stowage is located along the exterior sides of the module. A side hatch provides lunar egress and a standard berthing ring/hatch is located on the end for attachment to station. Four windows on the forward end provide viewing during lunar landing and a top window provides viewing for rendezvous and docking.

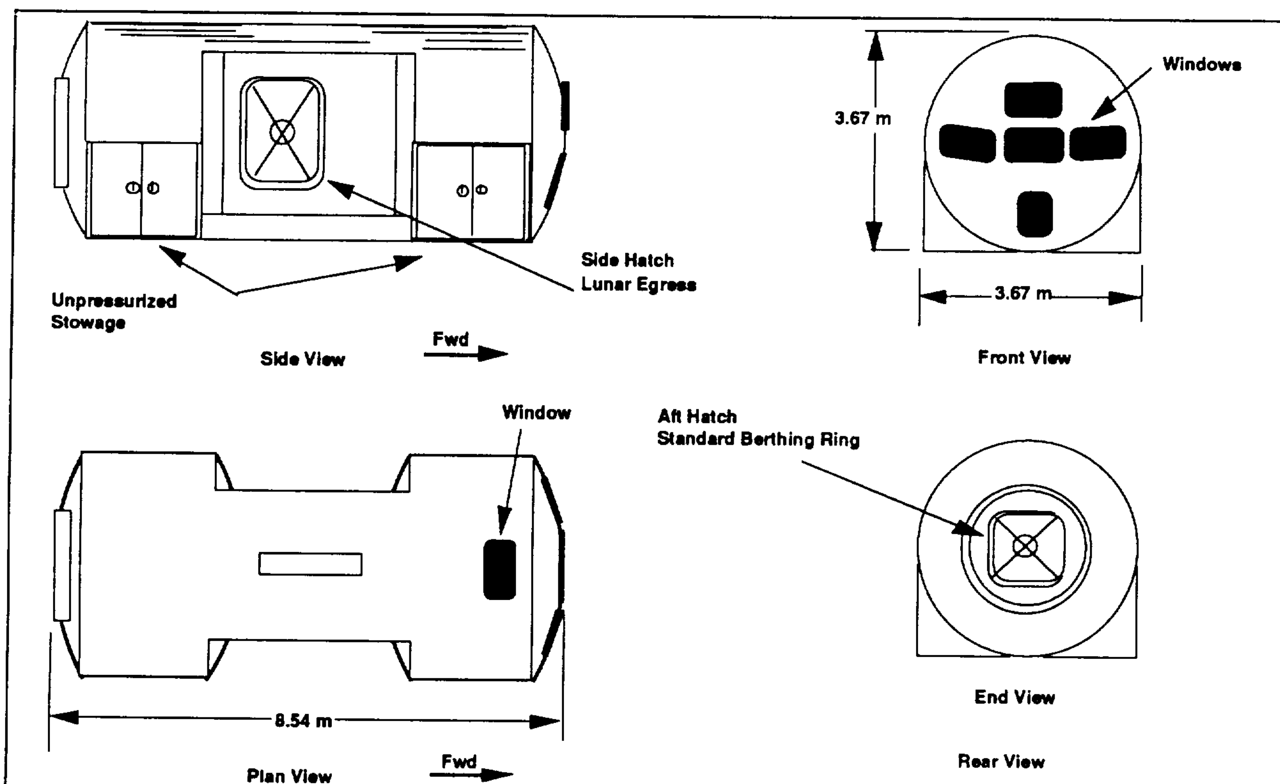


Figure 3.4.2-1 General Description of the Crew Module

The four unpressurized areas (Fig. 3.4.2-2) are provided to accommodate interface connections, stowage and ECLSS equipment. Two of the bays are designated for the avionics, power, and potable water interfaces between the core module and the crew module. These areas also house the batteries for backup power to the crew module. The other two bays are used to mount the cryo oxygen and nitrogen tanks need for the life support system. The advantage of these spaces is that they allow for outfitting and connecting the crew module to core module without having to enter the crew module during the assembly process. Thus, the vehicle is on the lunar surface the crew is able to check out the interfaces and avoid entering the crew module.

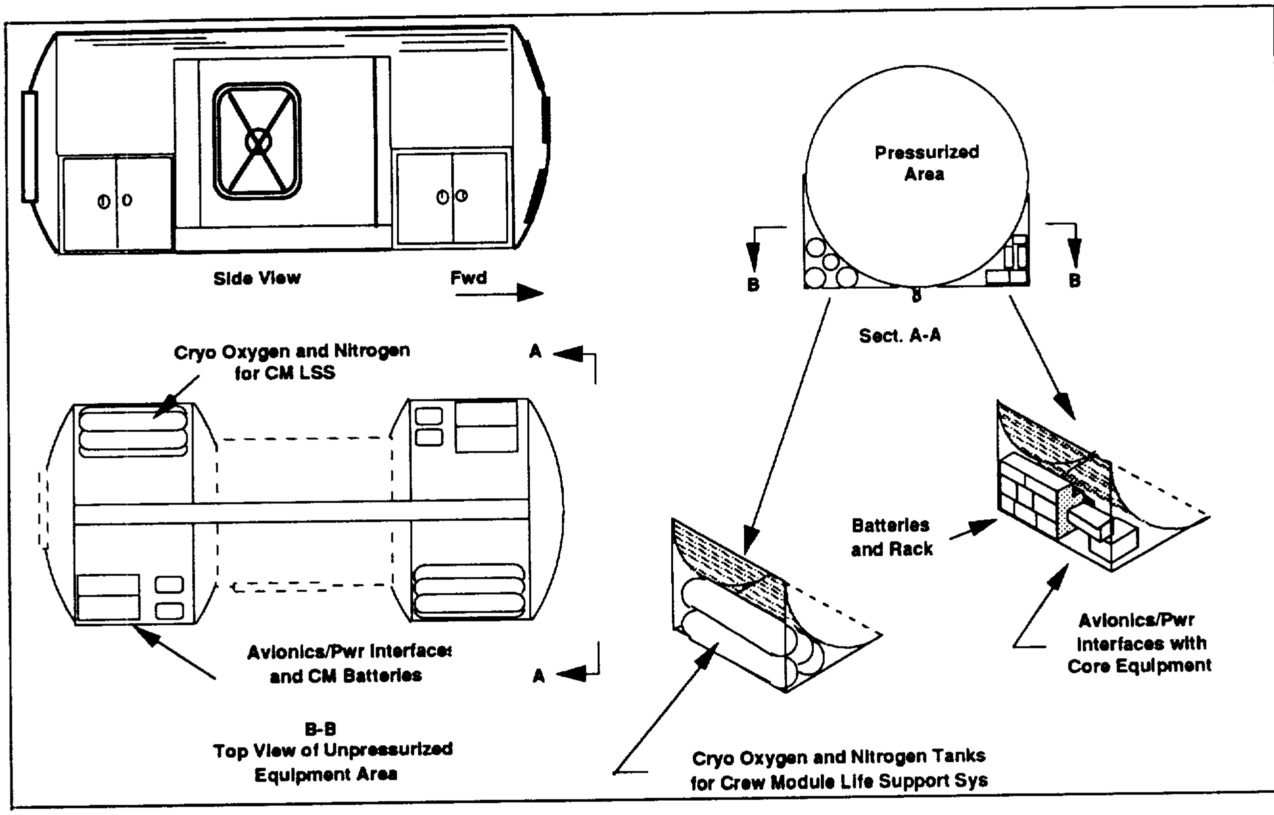


Figure 3.4.2-2 Unpressurized Areas

The layout of the crew module interior is shown in Figure 3.4.2-3. The starboard and port views of the interior show typical seating arrangements and the galley and waste management centers. The forward section houses the flight deck and seats three crewmen. The mid section provides stowage for four EMUs as well as providing lunar egress and storm protection. The aft section houses waste management and the galley and provides seating for one crewman. Equipment bays and internal stowage are located below the floor levels in all three sections. Light weight, portable, multipositional couches are used for sleep periods and body support during ascent, descent and aeropass.

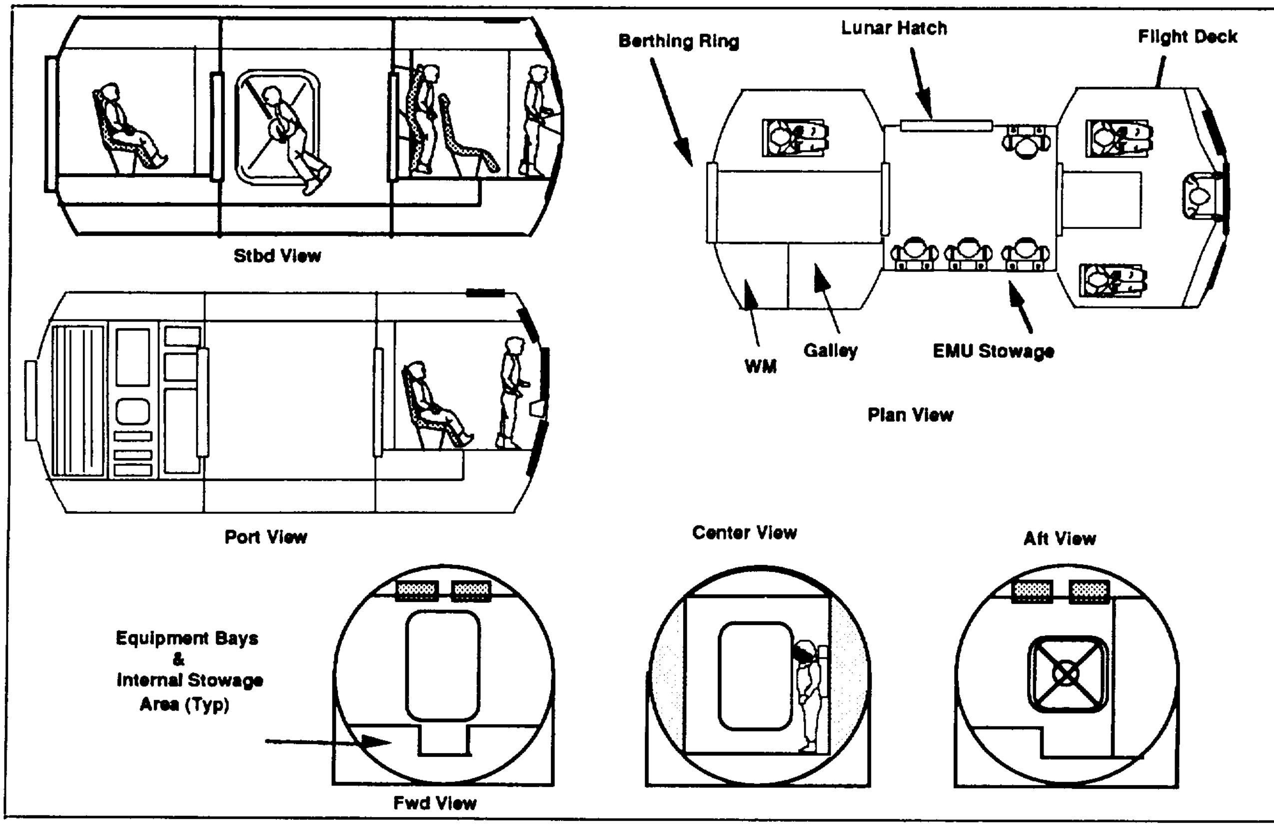


Figure 3.4.2-3 Crew Module Interior Layout

There are five windows which provide viewing for the crew. The four windows located in the forward end of the flight deck provide the pilot and co-pilot a field of view angle of over 170° for landing on the lunar surface (Fig. 3.4.2-4). The pilot also has a field of view angle of over 85° from the horizon to the lunar surface. A single window located in the top of the module provides the pilot with a view of the target during rendezvous and docking with the aerobrake in LLO. Figure 3.4.2-5 shows the crew arrangement during descent/ascent to/from the lunar surface and for LLO docking. The crew would wear their spacesuits during these operations in case there was a sudden depressurization of the module.

When the STV is ready to make the aeropass maneuver, the load forces felt by the crewmen are reversed from the normal acceleration force experienced throughout the mission. The crew would be in the wrong seating position and provisions had to be made to accommodate these load forces on the crew. Reentry couches, similar to those on the Apollo spacecraft, are mounted in the overhead. Prior to beginning the aeropass maneuver (Fig. 3.4.2-6), the crewmen would strap themselves into the reentry couches and thus be in the correct position for the aeropass loads. After the aeropass maneuver is completed, the crewmen would return to their normal seating position for circularization and rendezvous with SSF.

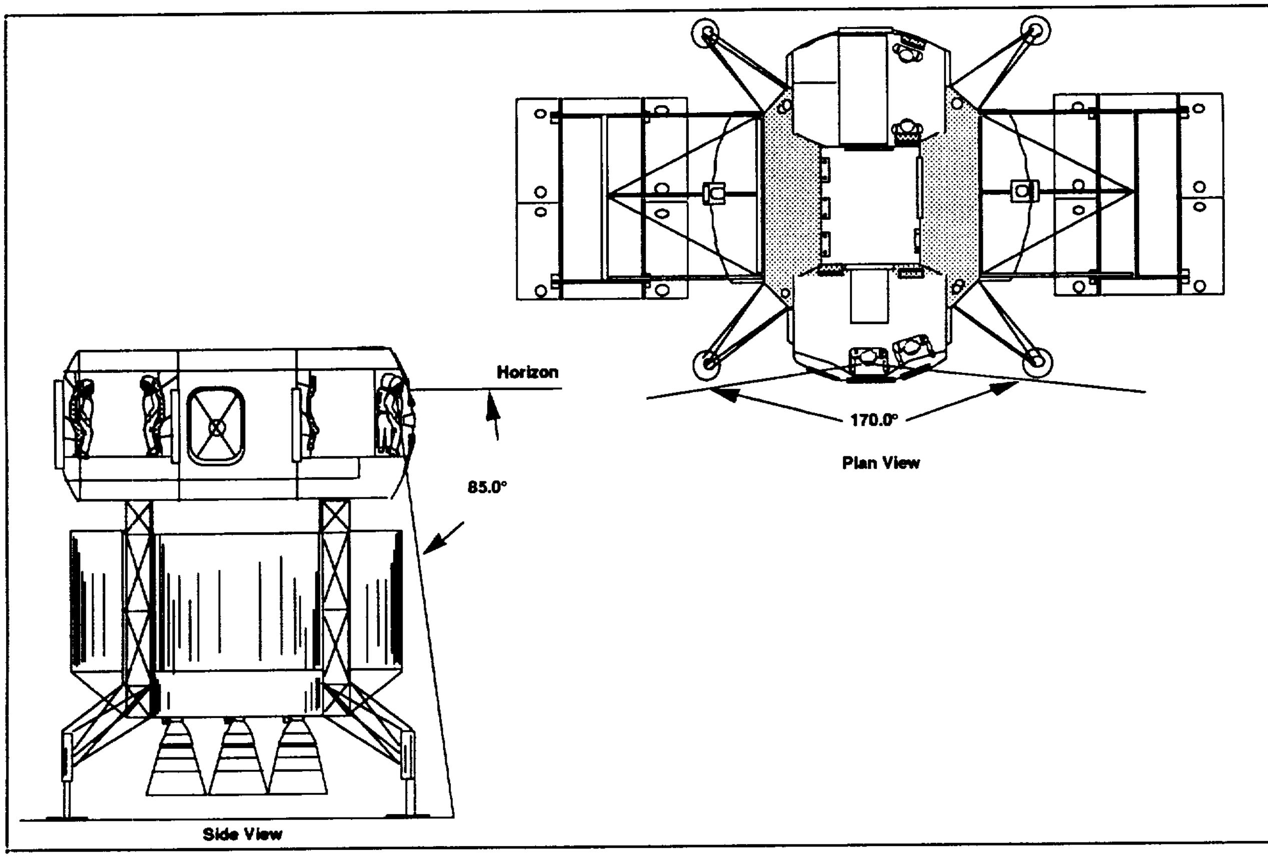


Figure 3.4.2-4 View Angle for Landing on the Lunar Surface

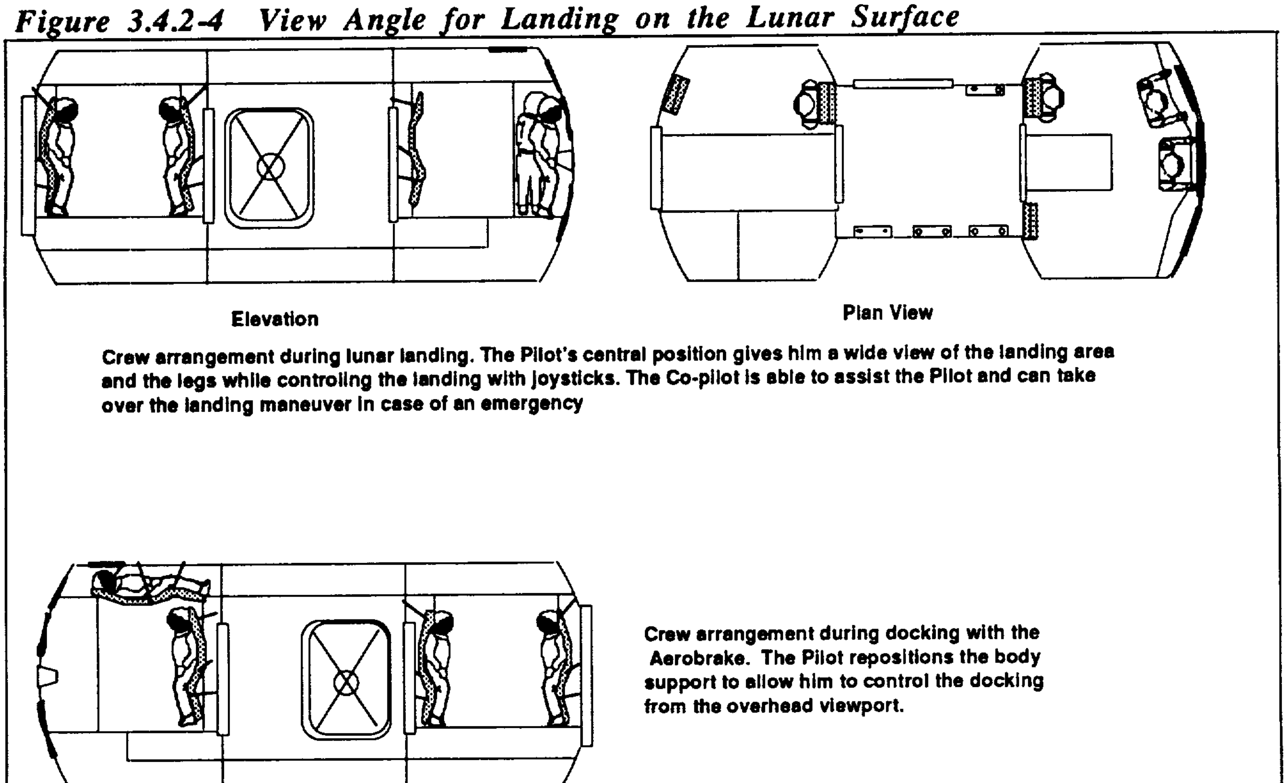


Figure 3.4.2-5 Crew Arrangement During Descent/Ascent

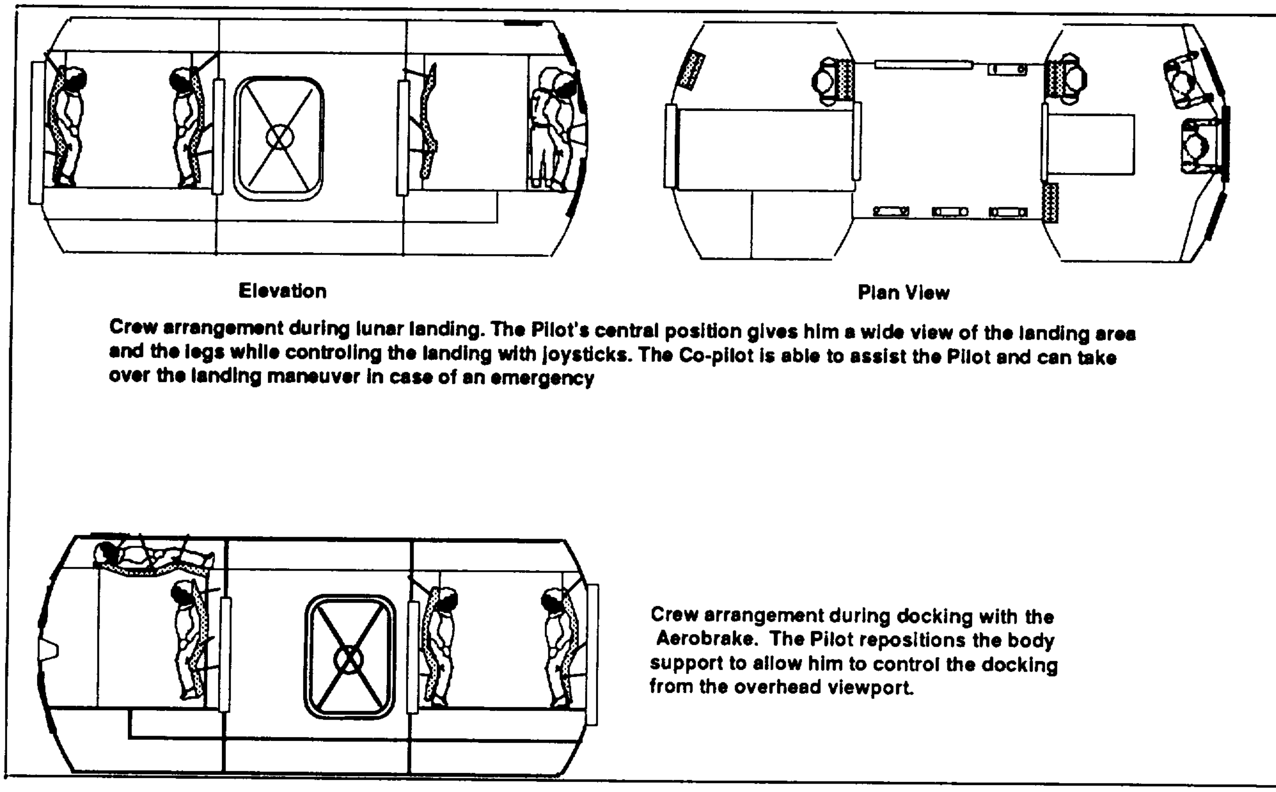


Figure 3.4.2-6 Crew Arrangement During Aeropass

In the event that a rescue mission is needed, the crew module can provide space for the additional crewmen. Two additional seat/reentry couches would be mounted in the mid section of the crew module. This will provide room for the rescue party, consisting of a pilot and co-pilot , and the four crewmen on the lunar surface to be rescued (Fig. 3.4.2-7). Table 3.4.2-1 gives the top level mass properties breakdown for the crew module.

3.4.3 Landing

After LTV has achieved LLO and stabilized its orbit, the crew prepares the vehicle for lunar descent. The aerobrake and the core separate and the core will back away from the aerobrake. The aerobrake will then deploy its solar array and assume a solar orientation. The crew then lowers the landing legs and checks to ensure that the legs are locked into places. The RCS thruster align the vehicle for the descent trajectory angle. Main engines are fired to brake the vehicle as it descends to the lunar surface. Once the vehicle has landed, the crew will checkout all the systems and prepare to disembark and offload the cargo.

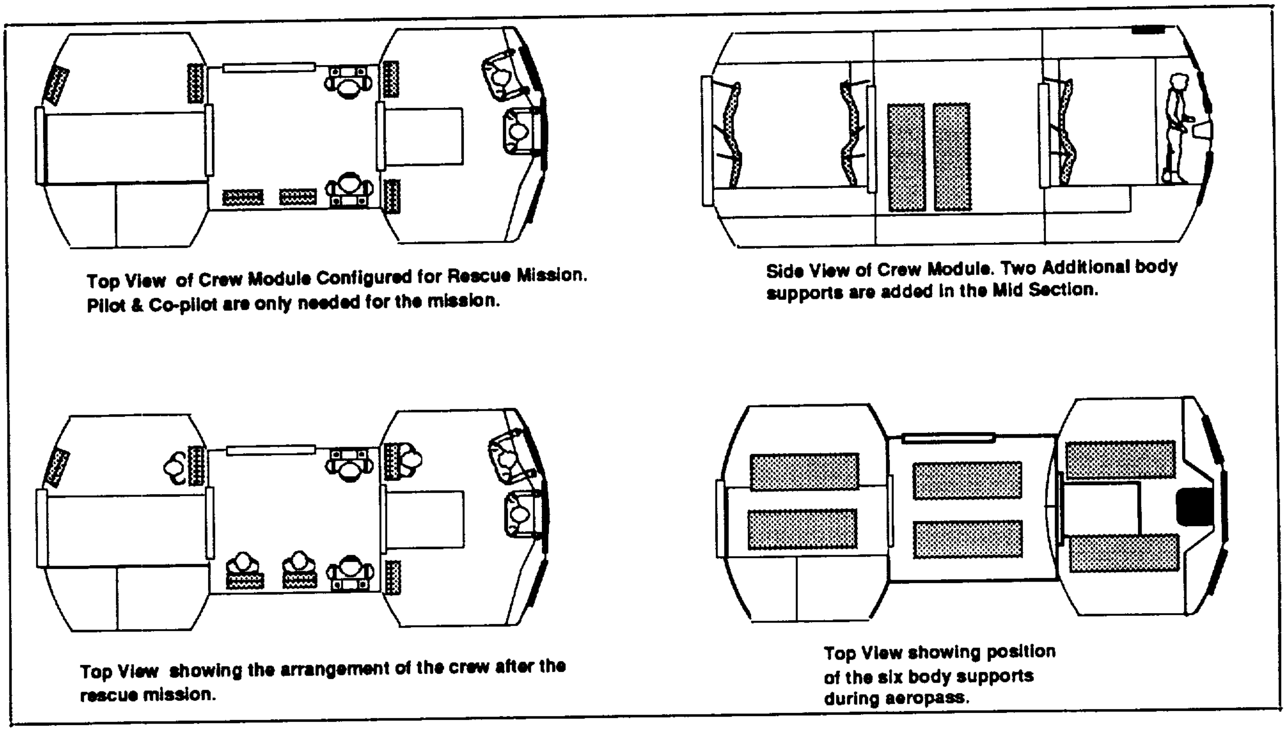


Figure 3.4.2-7 Crew Arrangement During Rescue Mission

Table 3.4.2-1 Mass properties Breakdown - crew module

Components	Mass (t)
Structure	3.32
ECLSS	1.52
Avionics/Power	0.87
Man Systems	0.17
Equipment & Spares	1.90
Total Dry Mass	7.78
Personnel & Consumables	0.66
Total Mass	8.44

3.4.4 Cargo Offloading

Cargo unloading of the piloted vehicle on the lunar surface can be accomplished without the use of the LEVPU. Once the vehicle has landed on the lunar surface (Fig. 3.4.4-1), the cargo can be lowered directly to the surface or onto a transporter by using a hoist mounted on the cargo support structure. The spacing between the legs of the core allow the cargo to be lowered directly to the surface.

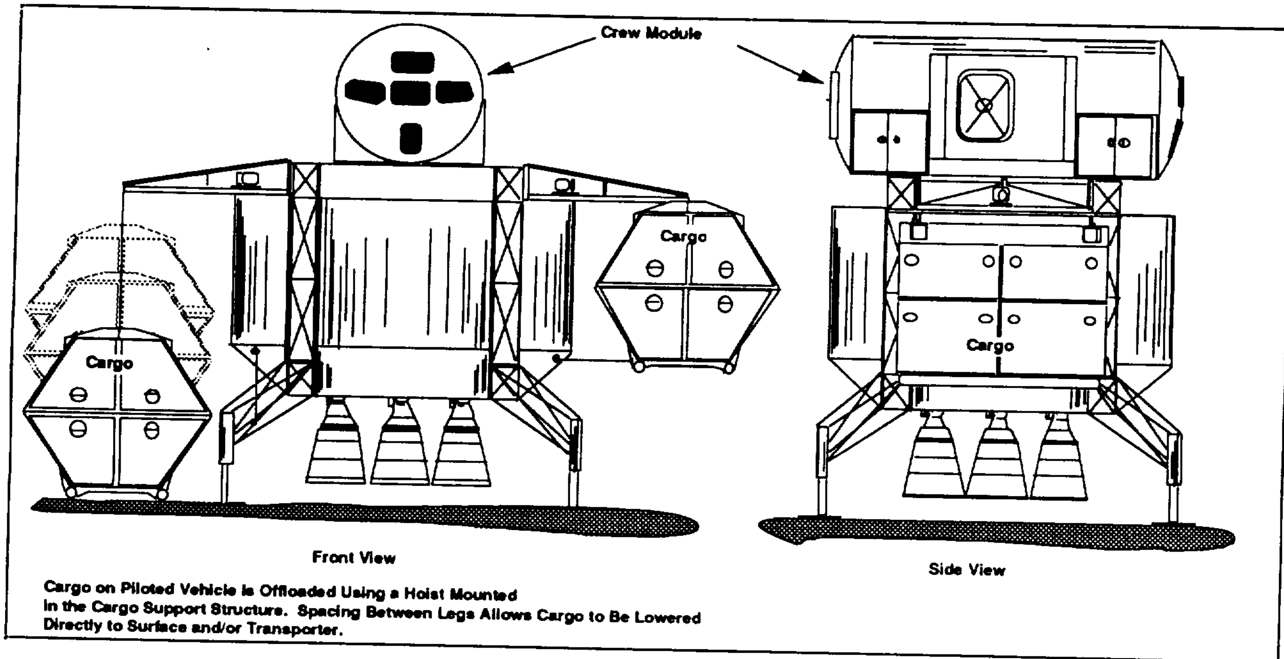


Figure 3.4.4-1 Piloted Vehicle Unloading Cargo On The Lunar Surface

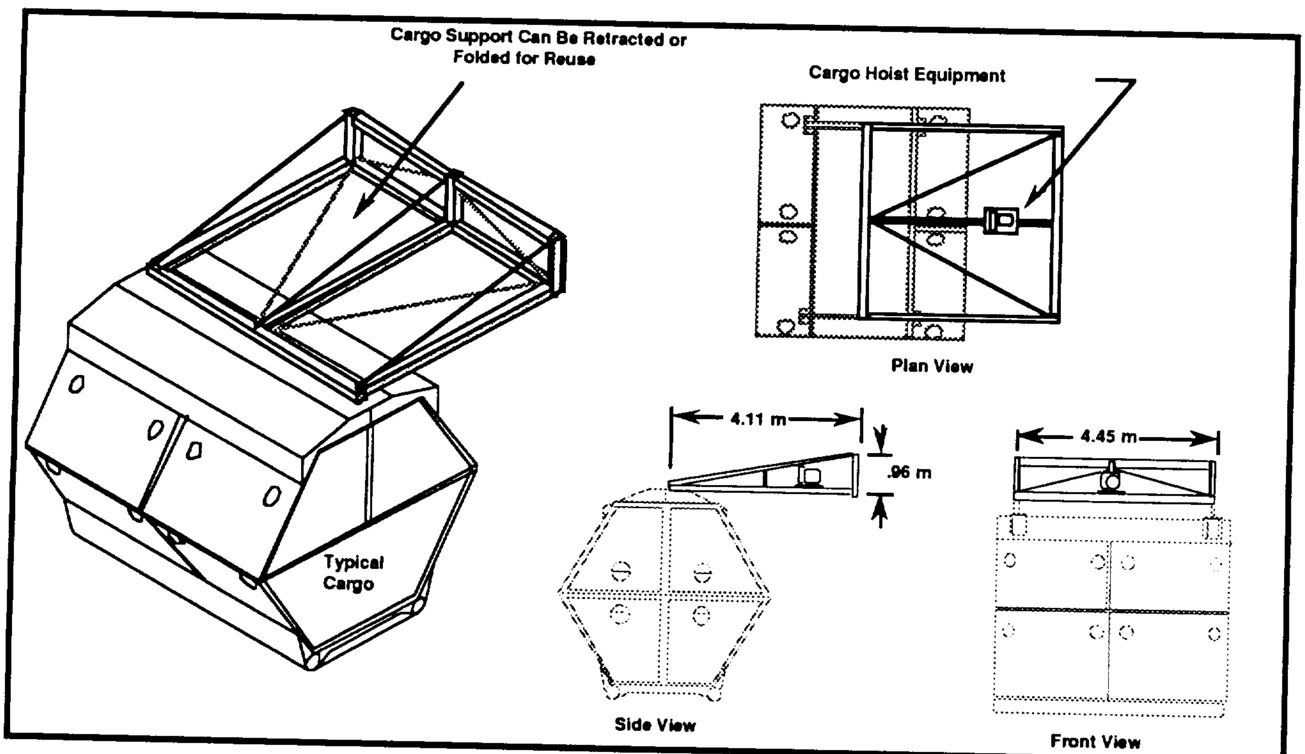


Figure 3.4.4-2 Piloted Vehicle With Cargo Following Landing

The cargo on the piloted configuration is supported by cargo supports (Figure 3.4.4-2) attached to each side of the core. The hoists located inside the cargo support structure allow the cargo to be lowered directly to the lunar surface. The structural dimensions of the cargo supports are shown.

These supports can be retracted or folded to fit within the aeroassist return configuration to allow reuse.

3.4.5 Rendezvous & Docking

After the core and crew module have lifted off from the lunar surface, they must rendezvous and dock in LLO with the aerobrake and its associated equipment for the return flight to SSF. The rendezvous procedure (Fig. 3.4.5-1) consist of aligning the two vehicles using a target located on the aerobrake. The docking probe on the crew module is extended and then engage with a grapple fixture located on the aerobrake. Guide rails located inside the aerobrake docking port will help align the vehicles. The docking probe will then be retracted pulling the crew module/core into the aeroassist position.

After the initial soft dock, the final docking procedure consist extending the four berthing mechanisms (Fig. 3.4.5-2) located on the upper platform of the core at each of the corners. These locking probes mate with receptacles located on the aerobrake. Once the final docking has been accomplished, two umbilical connections are made to transfer propellant from the return tanks located in the aerobrake to the engines in the core.

Details of the rendezvous and docking equipment are shown in Figure 3.4.5-3. The aerobrake docking fixture is located in the center of the aerobrake and consists of a grapple fixture with an end effector with range of ±15°. The grapple fixture is retracted into the fixture once docking has occurred. Three guide rails placed at 120° around the fixture help align the vehicles. Also shown is the berthing mechanism used for making the firm attachment between the vehicles. The four berthing probes extend up from the core and engage in the berthing mechanisms located in the aerobrake along the bulkhead.

3.4.6 Return Configuration

After the crew module and propulsion/avionics core has ascended from the lunar surface and rendezvoused and docked with the aerobrake/equipment in LLO, the crew module, core, and aerobrake are returned to SSF using the propellants in the return tanks located in the aerobrake. The piloted return configuration and CGs at the beginning of the aeropass are shown in Figures 3.4.6-1 and 3.4.6-2 respectively. Once the landing legs of the core are retracted, the crew module and core fit within the 22° wake angle of the aerobrake for the aeroassisted return. The total return mass leaving LLO is approximately 27 t.

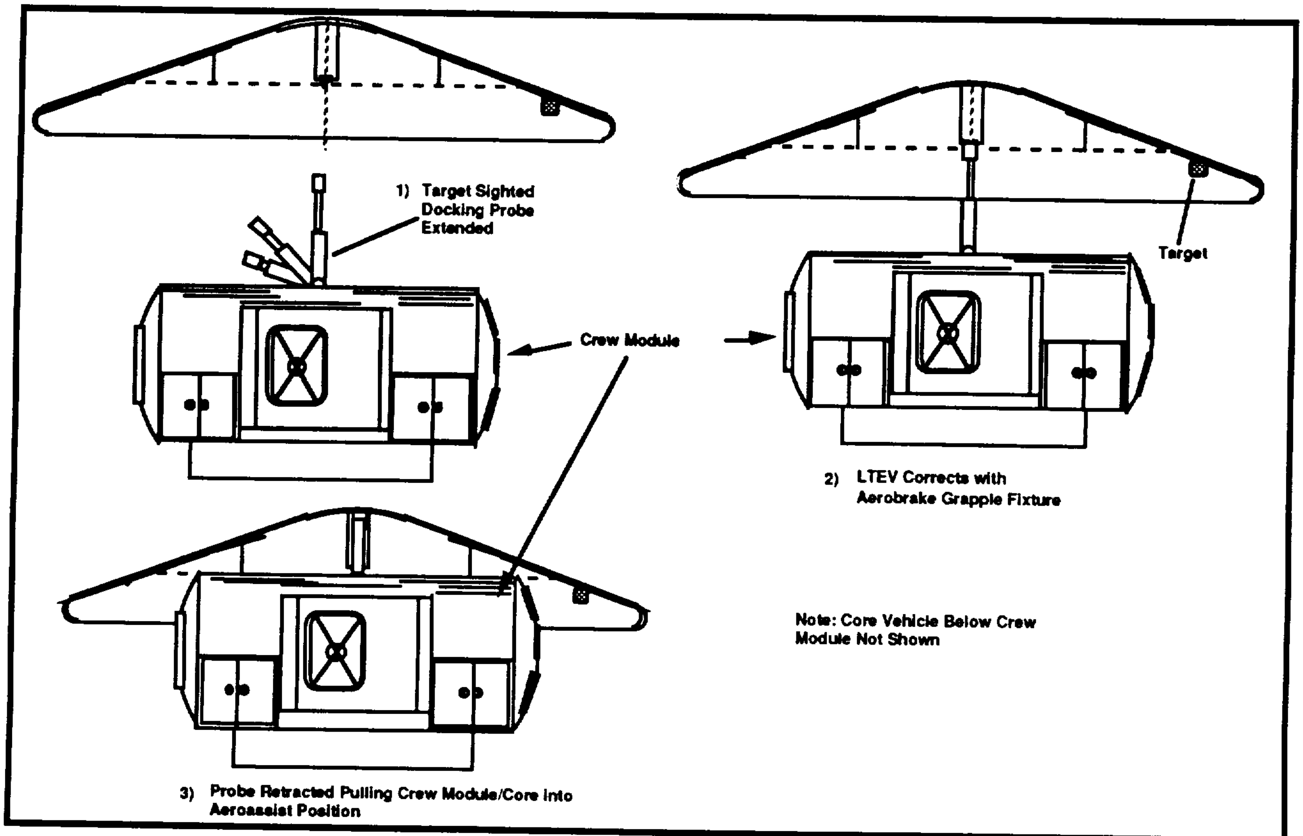


Figure 3.4.5-1 Rendezvous Procedure

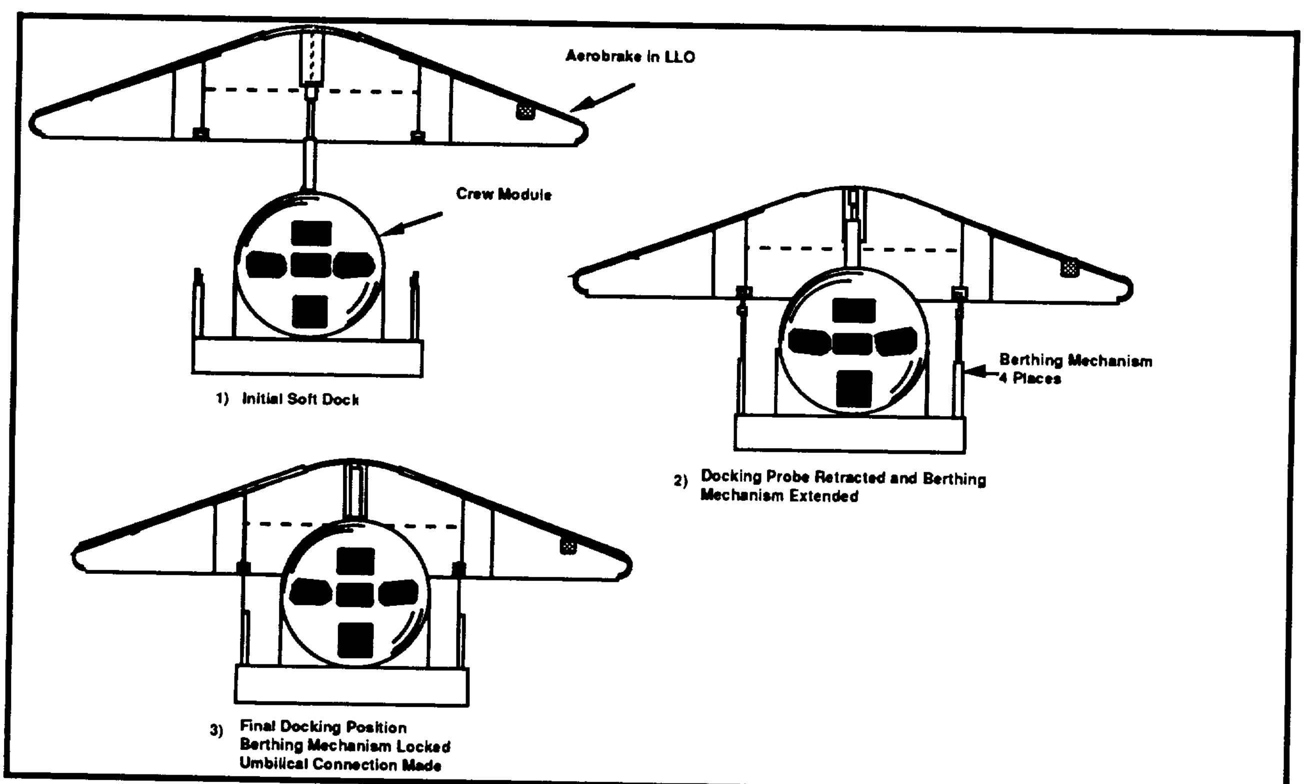


Figure 3.4.5-2 Berthing Mechanisms

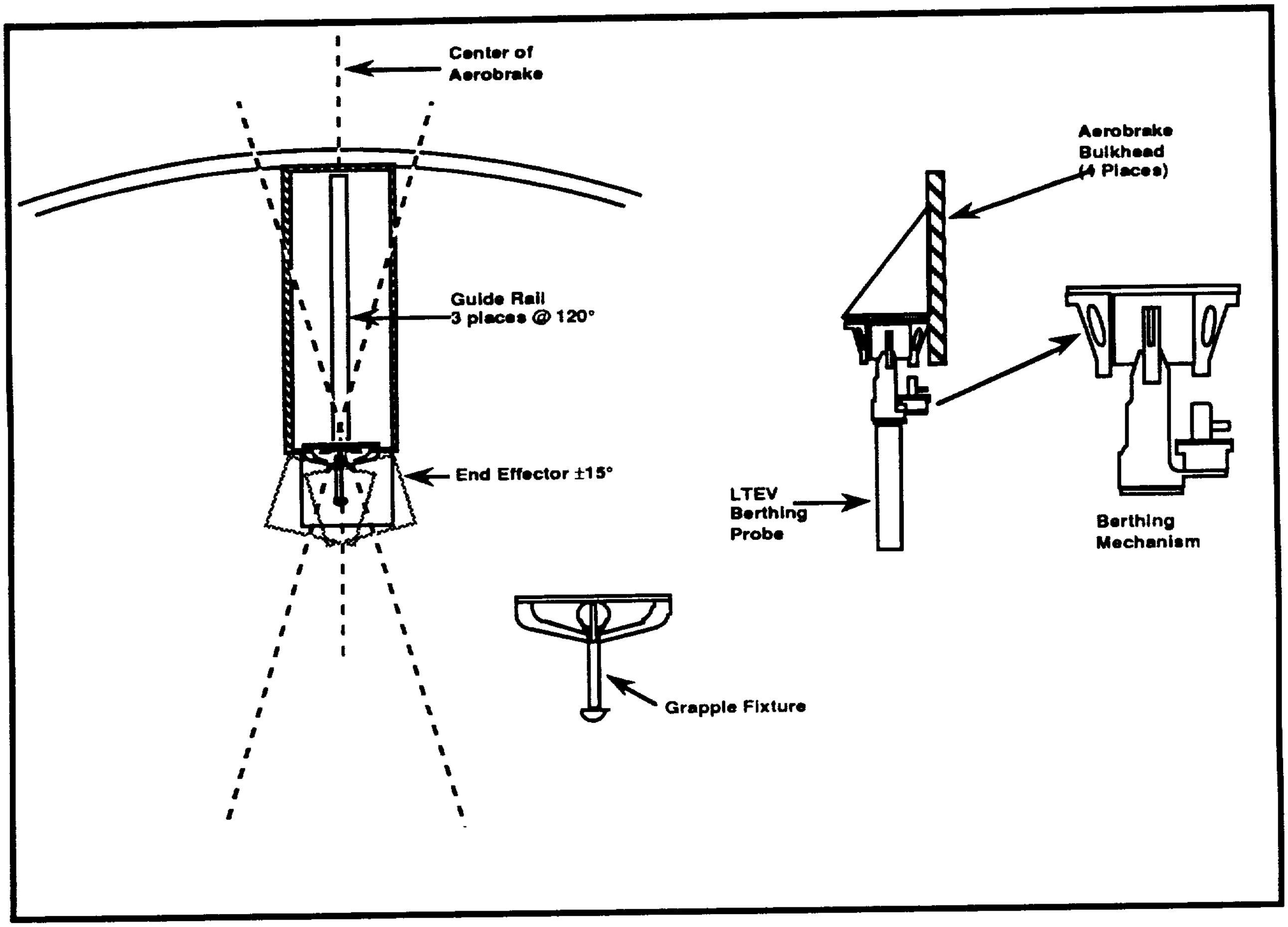


Figure 3.4.5-3 Details of the Rendezvous and Docking Equipment

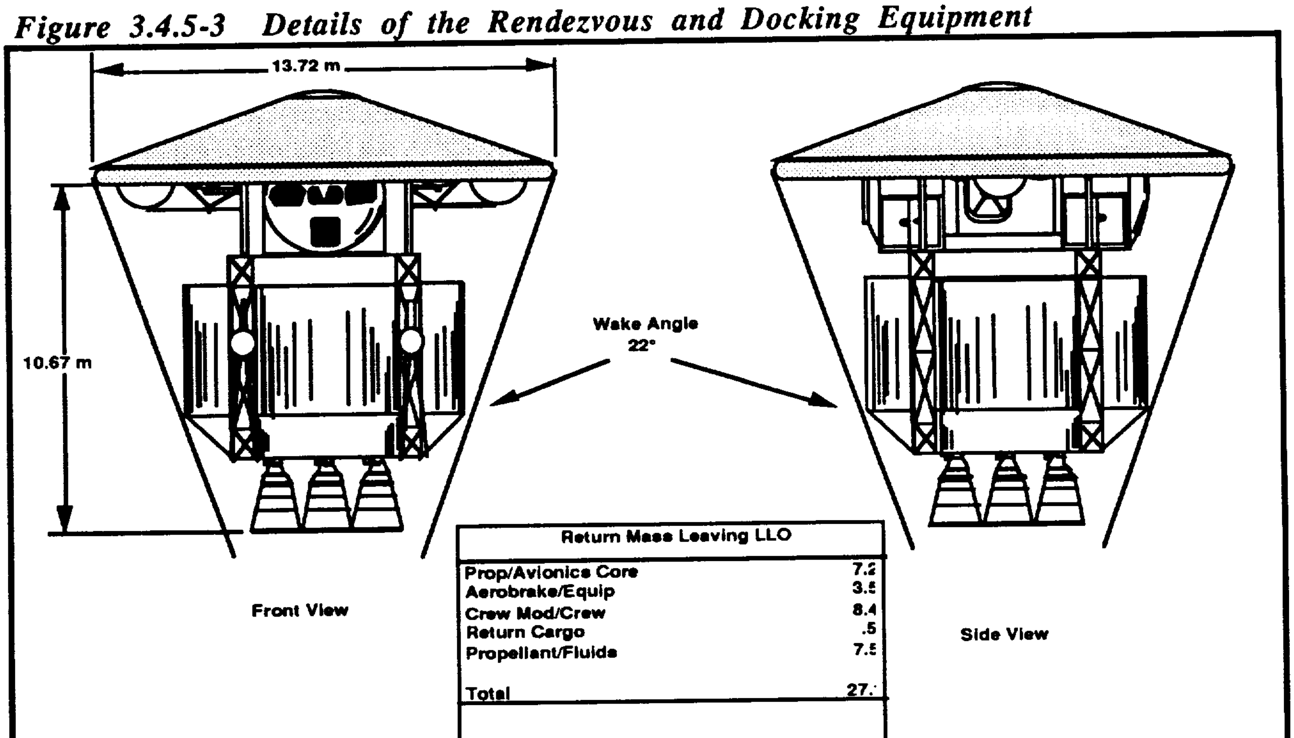


Figure 3.4.6-1 Piloted Return Configuration and CGs at the Beginning of Aeropass

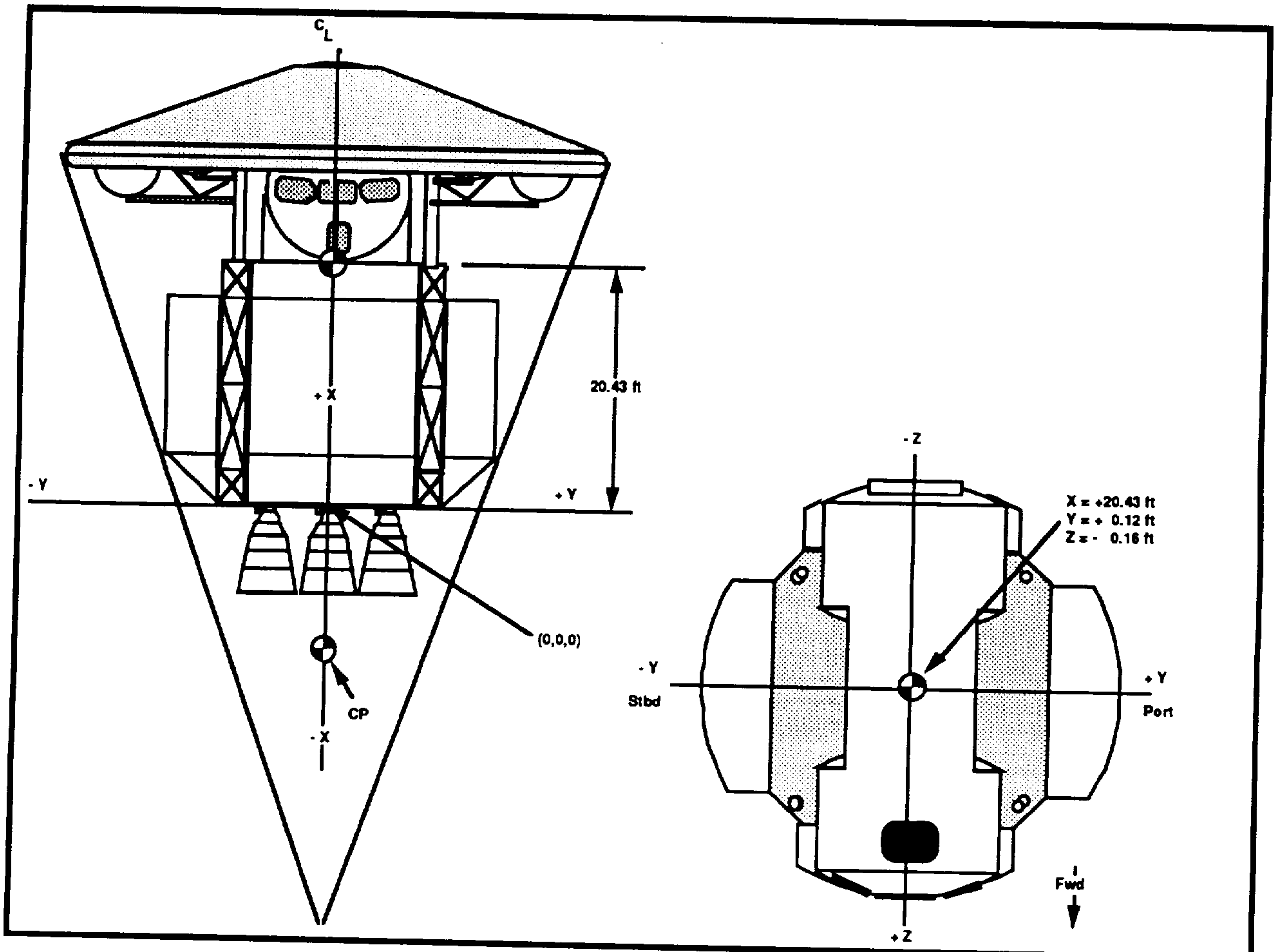
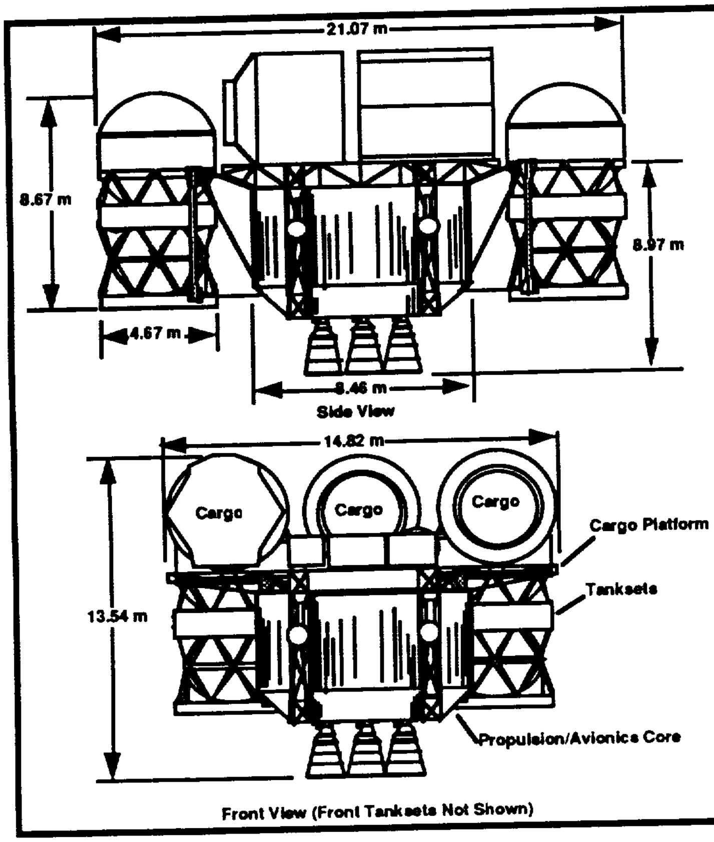


Figure 3.4.6-2 Piloted Return Configuration and CGs at the Beginning of Aeropass

3.5 Cargo Configuration

The cargo configuration is composed of the propulsion/avionics core, a large structural platform, and the drop tanksets common to the piloted configuration. It is designed to deliver 33 mt to the lunar surface in an expendable mode. Figure 3.5-1 shows the overall dimension of the vehicle as it prepares to leave from LEO. The vehicle is 13.54 m (including the height of the payload) by 14.82 m by 21.07 m. The drop tanks are extended two meter out further then the piloted vehicle to accommodate the width of the platform. Core will provide minimum interfaces to the cargo - power but no thermal control. The propellant requirement for the cargo missions is lower than that required for a piloted mission. To keep commonality between both configurations, the drop tanks as stated before are the same as those on the piloted vehicle, however propellant is offloaded to meet the mission requirements. The vehicle can deliver up to 37.4 mt of cargo.



Mass Properties	
Components	Mass (t)
Propulsion/Avionics Core	7.1
Tanksets (4 TLI & 2 LOI)	9.1
Cargo Platform	2.4
Vehicle Dry Mass	18.7
Propellant Mass	146.5
Cargo Mass	33.0
Total Mass	198.2

Figure 3.5-1 Overall Dimension of Vehicle Leaving LEO

3.5.1 Mass Properties

Table 3.5.1-1 gives the top level mass properties of the cargo configuration at ignition as it is ready to leave from LEO.

3.5.2 Cargo Platform

In order to accommodate the large volume cargo manifested to the lunar surface, special structure must be added to the basic core structure to provide structural support. The overall view of the platform is shown in an isometric view (Fig. 3.5.2-1) with the structural elements of the core vehicle. The cargo support area (Fig. 3.5.2-2) is approximately 14.8 m by 10.5 m in size once all the cargo extensions have been added. The larger area is formed by adding two central platform extensions and two outer platform extensions to the basic core structure. These extensions are made of lightweight truss work and can folded and returned for additional uses. Cargo is mounted using center keel and trunnion fittings similar to those on the STS.

Table 3.5.1-1 Mass Properties Breakdown - Cargo Configuration

	DESCRIPTION	MASS	MASS	MASS
	CARGO SUMMARY AT IGNITION	KG	KG	M.TONS
02	STRUCTURE		2363.15	2.36
03	PROPELLANT TANKS		8721.84	8.72
04	PROPULSION SYSTEM		380.34	0.38
05	MAIN ENGINES		1150.11	1.15
06	RCS SYSTEM		122.45	0.12
07	G. N. & C.		195.46	0.20
08	COMMUNICATION & DATA HNDLG		242.70	0.24
09	ELECTRICAL POWER		444.22	0.44
10	THERMAL CONTROL SYSTEM		553.47	0.55
11	AEROBRAKE		0.00	0.00
19	GROWTH		2440.00	2.13
	DRY WEIGHT		16299.80	16.30
12	CARGO PLATFORM		2450.00	2.45
15	PROPELLANTS		146563.04	146.56
01	CORE TANKS	23900.00		
02	DROP TANKS TLI	102500.00		
03	DROP TANKS LOI	20100.00		
04	RETURN TANKS	0.00		
05	FLUIDS & PRESSURANTS	63.04		
17	PERSONNEL		0.00	0.00
18	CARGO		33000.00	33.00
	TOTAL WEIGHT		198312.84	198.31

3.5.3 Lunar Manifest

Figure 3.5.3-1 shows an isometric view of the payload manifested for the second cargo expendable flight (designated Flight 1). The cargo consists of the lunar habitat module, airlock, a power module and one cargo pallet. Total manifested mass for this flight is 26.3 mt. There are four designated cargo flights and the mass cg as defined in the PSS documents has been laid out to aid in cg control for flight and landing. The four cargo flight manifests are detailed in Figure 3.5.3-2, with the total manifest mass given for each flight. Cargo Flight 0 will deliver the LEVPU, a three legged crane that will unload all the other cargo flights and can assist in unloading the cargo from the piloted vehicle if required. The LEVPU is designed to be self unloading.

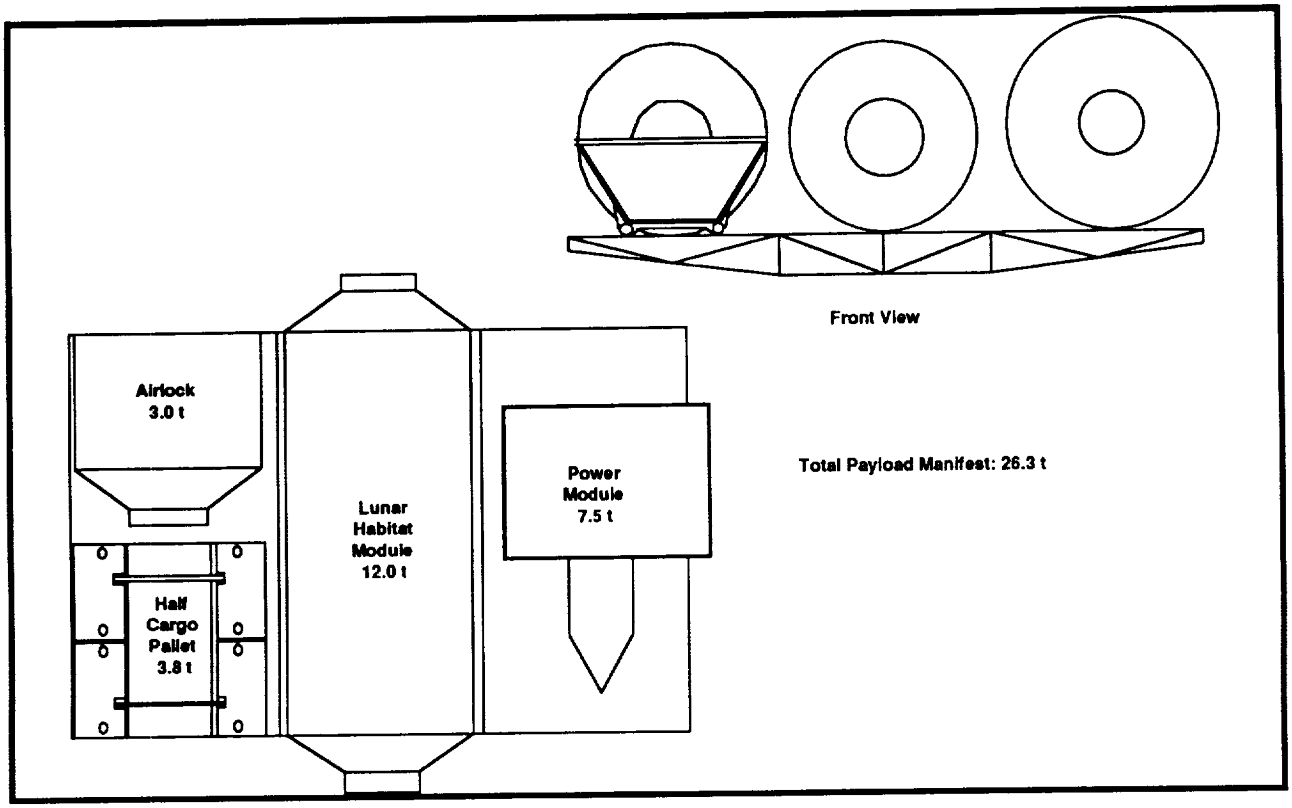


Figure 3.5.2-1 Cargo Platform Isometric View

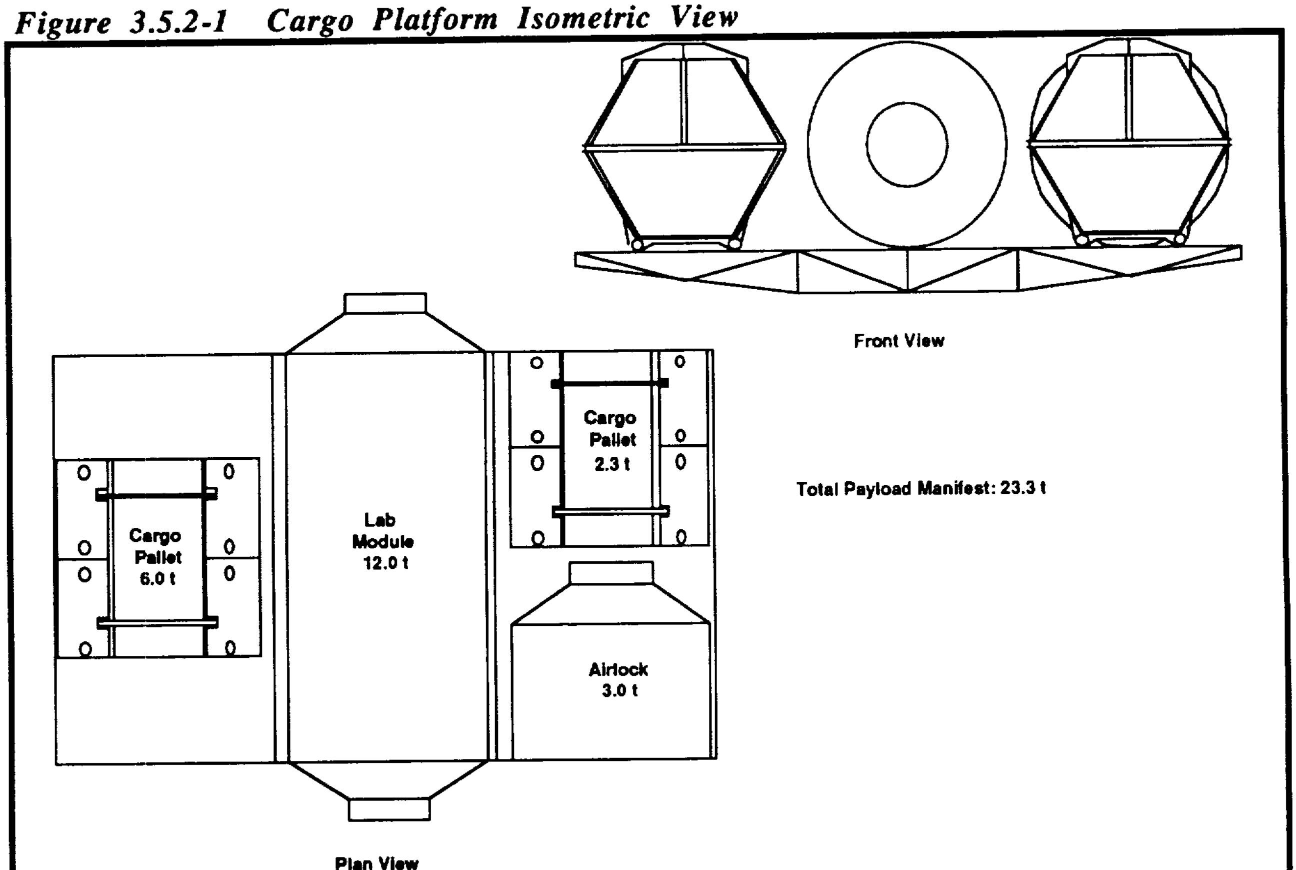


Figure 3.5.2-2 Cargo Support Area

3.5.4 Cargo Offloading

Figure 3.5.4-1 shows how the LEVPU will unload the cargo from the cargo expendable configuration once the vehicle has landed on the lunar surface. The platform and the vehicle size allows the payload unloader to roll over and straddle the vehicle with its cargo. Once positioned over the vehicle the unloader picks up a piece of cargo, lifts it, and proceeds to roll away from the vehicle. After the cargo has been deposited in its position on the lunar surface or on a transporter, the unloader will proceed back to the vehicle to unload another piece of cargo.

3.6 Cargo Reusable Configuration

An optional cargo reusable configuration (Figs. 3.6-1 and 3.6-2) for the single propulsion system concept has been proposed. The six tanksets, an aerobrake and the large cargo platform are attached to the common propulsion/avionics core. The four docking probes shown on the piloted vehicle can be position to accommodated the larger payload heights. The configuration can deliver approximately 26 tonnes of cargo to the lunar surface and return the vehicle to SSF using 169.3 tonnes of LO2/LH2 propellant. The 13.72 m rigid aerobrake protects the vehicle during the aeroassisted return to SSF.

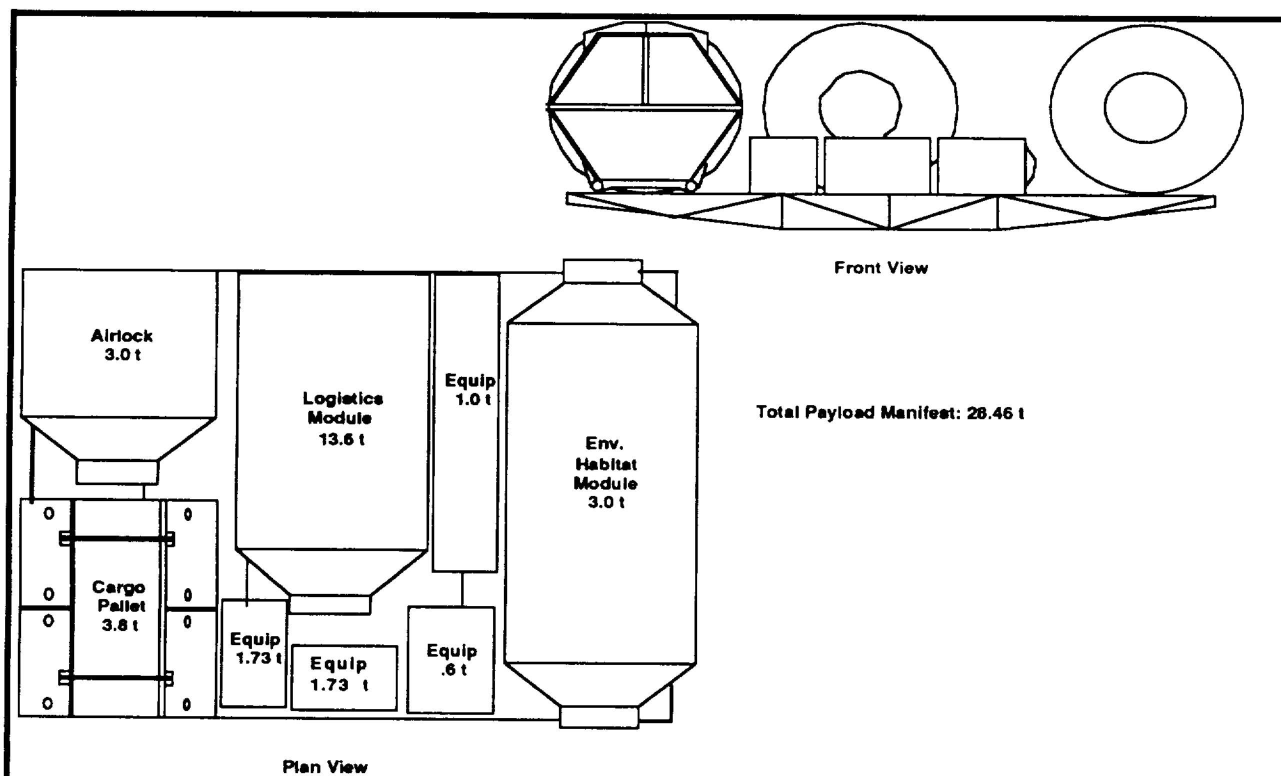


Figure 3.5.3-1 Isometric View Of Payload (Flight 8)

C-4

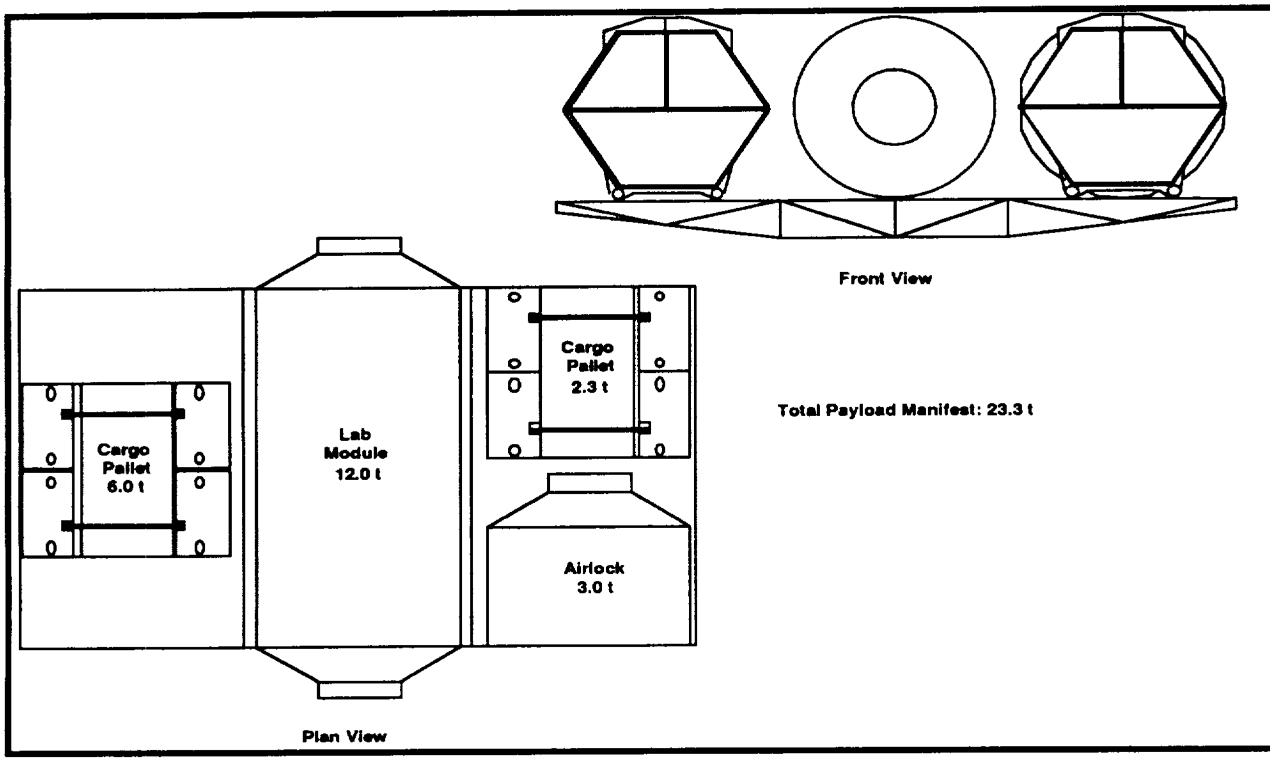


Figure 3.5.3-2 Detailed Cargo Flight Manifest (Flight 4)

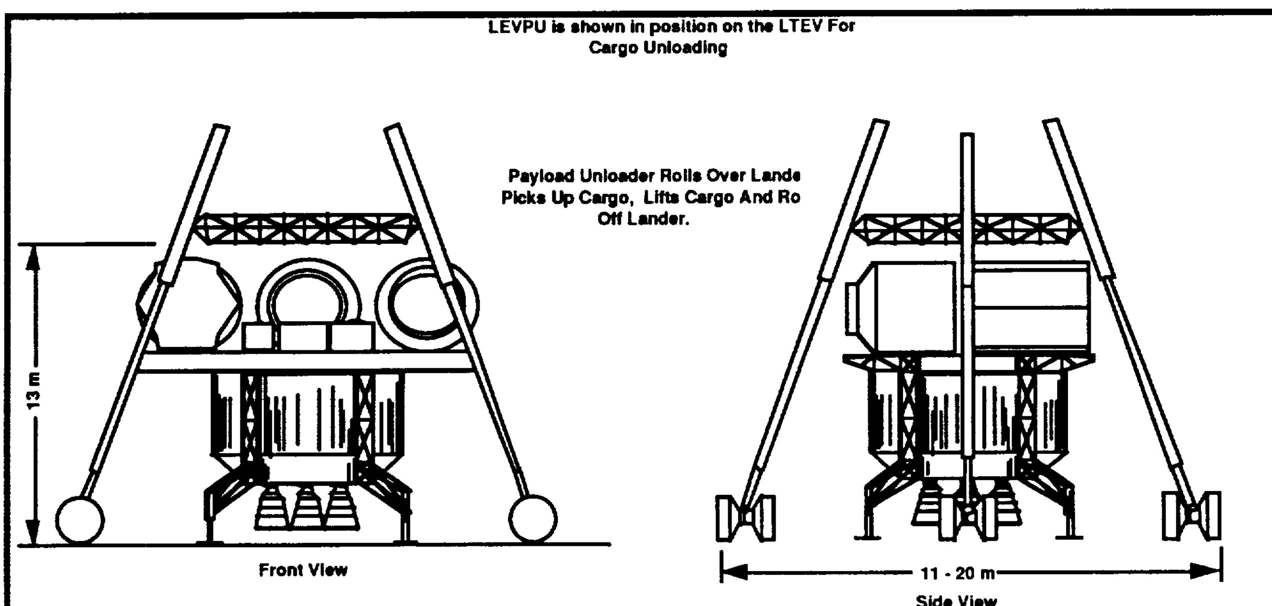


Figure 3.5.4-1 Shows LEVPU Unload Cargo

3.6.1 Mass Properties

Table 3.6.1-1 gives the top level mass properties breakdown of the cargo return vehicle at ignition as it ready to leave from LEO.

Table 3.6.1-1 Mass Properties Breakdown - Cargo Return Configuration

	CARGO SUMMARY AT IGNITION	KG	KG	M.TONS
02	STRUCTURE		2363.15	2.36
03	PROPELLANT TANKS		8721.84	8.72
04	PROPULSION SYSTEM		380.34	0.38
05	MAIN ENGINES		1150.11	1.15
06	RCS SYSTEM		122.45	0.12
07	G. N. & C.		195.46	0.20
08	COMMUNICATION & DATA HNDLG		242.70	0.24
09	ELECTRICAL POWER		444.22	0.44
10	THERMAL CONTROL SYSTEM		553.47	0.55
11	AEROBRAKE		2070.00	2.07
19	GROWTH		2440.00	2.44
	DRY WEIGHT		18676.00	18.68
12	CARGO PLATFORM		2450.00	2.45
15	PROPELLANTS		169300.00	169.30
17	PERSONNEL		0.00	0.00
18	CARGO		26000.00	26.00
	TOTAL WEIGHT		198312.84	213.98

3.6.2 Rendezvous & Docking

The rendezvous and docking of the return cargo vehicle is similar to the procedure described in the piloted section. The operations are combination of automatic controls and radio controls from either the SSF or earth. Since the crew module is not present, a longer docking probe must be attached to the core module to make the initial soft docking with the orbiting aerobrake.

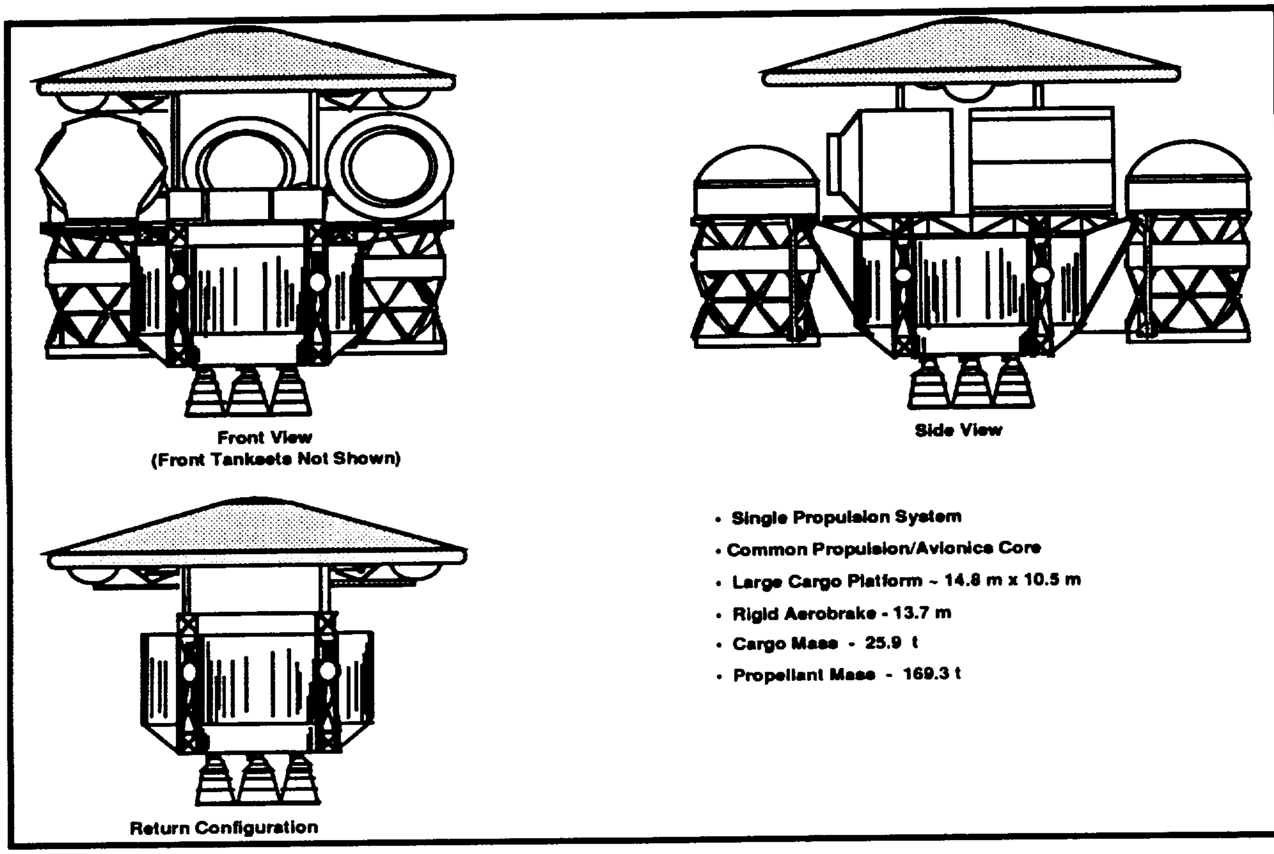
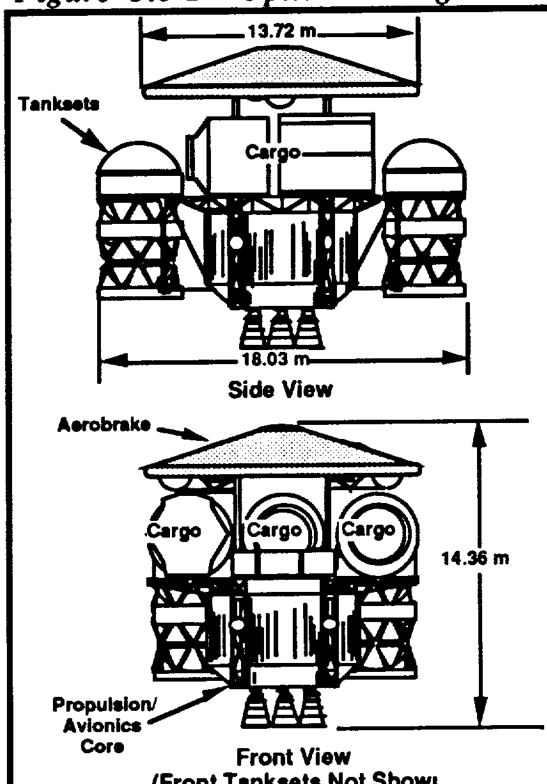


Figure 3.6-1 Optional Cargo Reusable Configuration Overview



Mass Properties	
Components	**Mass (t)**
Prop/Avionics Core	7.19
Tanksets (4 TLI & 2 LOI)	9.11
Cargo Platform	2.45
Aerobrake & Equip	3.50
Vehicle Dry Mass	22.25
Propellant	169.3
Cargo	25.9
Total Mass	217.45

Figure 3.6-2 Optional Cargo Reusable Configuration Summary

3.6.3 Return Configuration

The cargo return vehicle must meet the same requirements for the aeropass maneuver as the piloted vehicle. Figure 3.6.3-1 shows the vehicle layout as it begins to make the aeropass maneuver. The vehicle is within the prescribed wake angle of 22°. The overall vehicle dimensions are 13.72 m by 10.67 m.

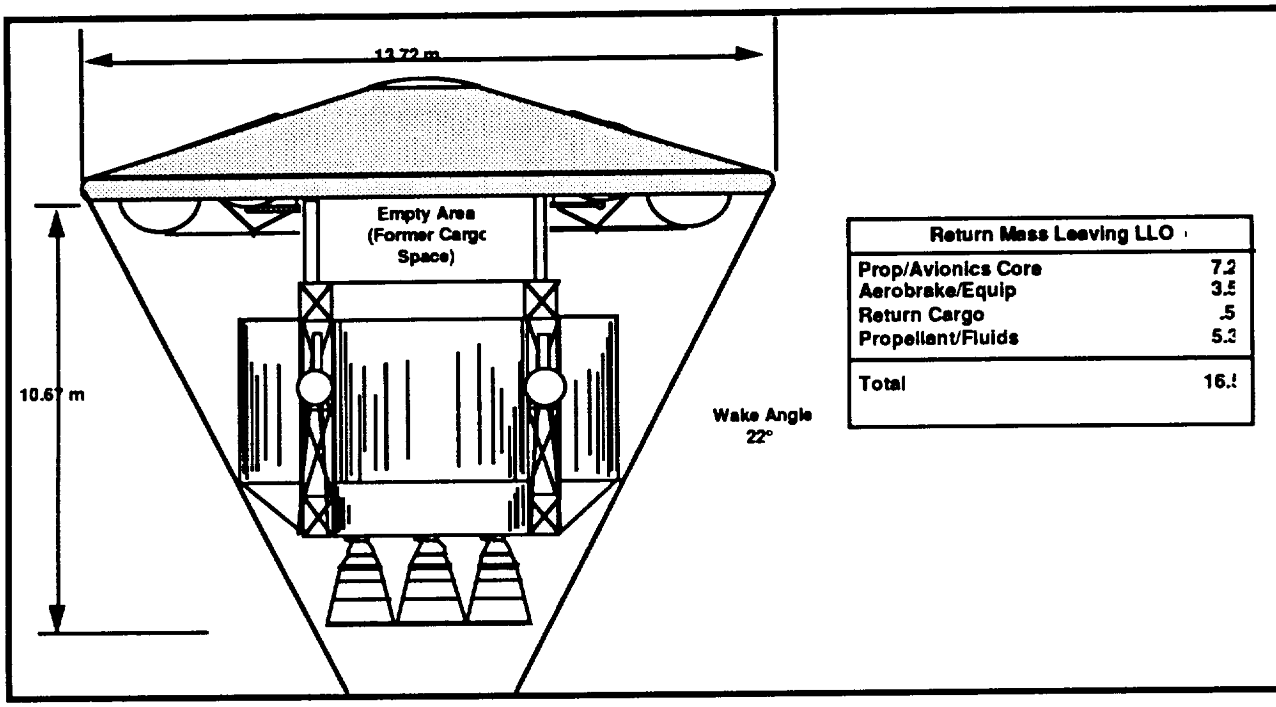


Figure 3.6.3-1 Vehicle Layout Beginning Aeropass Maneuver

3.7 Initial & Growth STV Concept Definition

A common set of engines, tanksets, cores, aerobrakes, crew modules, subsystems, etc., were found to be applicable in the development of various ground- or space-based, expendable or reusable STV configurations including the lunar transportation system.

The ability of the baseline vehicle or elements of the baseline vehicle to perform the other DRM cargo requirements was evaluated and is depicted in Table 3.7.1-1. All DRM cargo requirements can be met by either the initial STV or the baseline's core vehicle with only one set of drop tanks. The capability of the stages was determined using the RL10A-4 cryogenic engine at 449.5 seconds of Isp and the various pieces of the LTV as listed in table. The table shows the minimum needs of the core vehicle to meet the DRM cargo requirements in terms of extra propellant and subsystems, e.g., the crew module for the manned mission.

Table 3.7.1-1 Baseline Vehicle Adaptability

DRM	Description	Cargo Requirement	LTS/STV Configuration
E - 1	Manned GEO Servicing	4.0 t delivery & return	4E-5B Core w/AB, Crew Module, & 43 t Prop in Drop Tanks
E - 2	10 t GEO Platform Delivery *(DELETED IN CNDB '90)*	10.0 t delivery	Interim Vehicle (12.9 t maximum capability)
E - 3	6.4 t GEO Payload Delivery (DoD)	6.4 t delivery	Interim Vehicle (12.9 t maximum capability)
E - 4	Unmanned Polar Platform Servicing	3.5 t delivery & return	4E-5B Core w/AB, & 26.3 t Prop in Drop Tanks
P - 1	Comet Nucleus Sample Return *(DELETED IN CNDB '90)*	16.0 t delivery	4E-5B Core & 5.1 t Prop in Drop Tanks

DRM Propellent Loads Are Based on the Use of RL10A-4 Engines (449.5 sec)

3.7.1 Expendable Initial Concept

The initial STV, a ground-based expendable version, can be built from the common set of elements and subsystems (Fig. 3.7.1). A common tankset and two engines with limited subsystems form the basis for this vehicle. It is sized to fit within a 4.6 m (15 ft) diameter payload shroud for delivery to orbit. The dry weight of the vehicle is about 3 t with a length of nearly 12 m. With approximately 28 tonnes of LO2/LH2 propellant in the tankset, the vehicle can deliver 12.9 tonnes of payload to a geosynchronous orbit.

3.7.1.1 Mass Properties

Table 3.7.1.1 gives the top level mass properties breakdown for the ground-based, expendable STV.

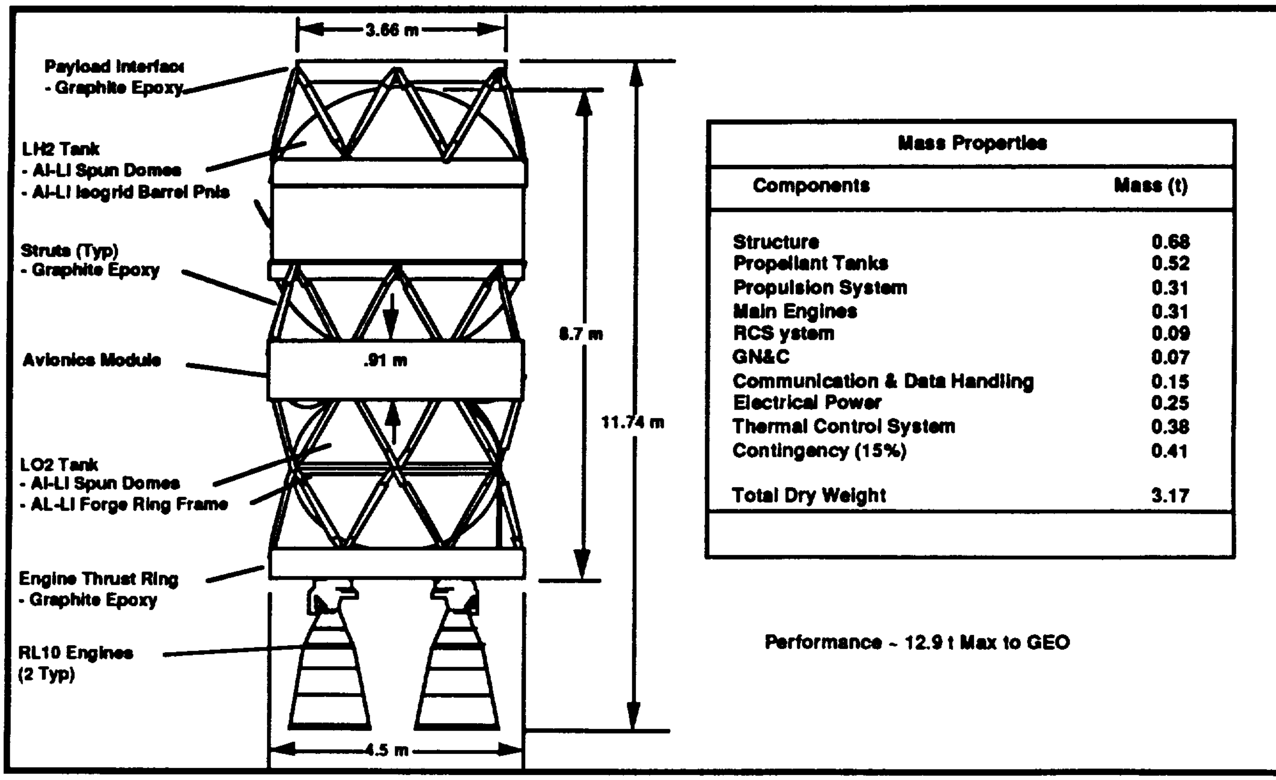


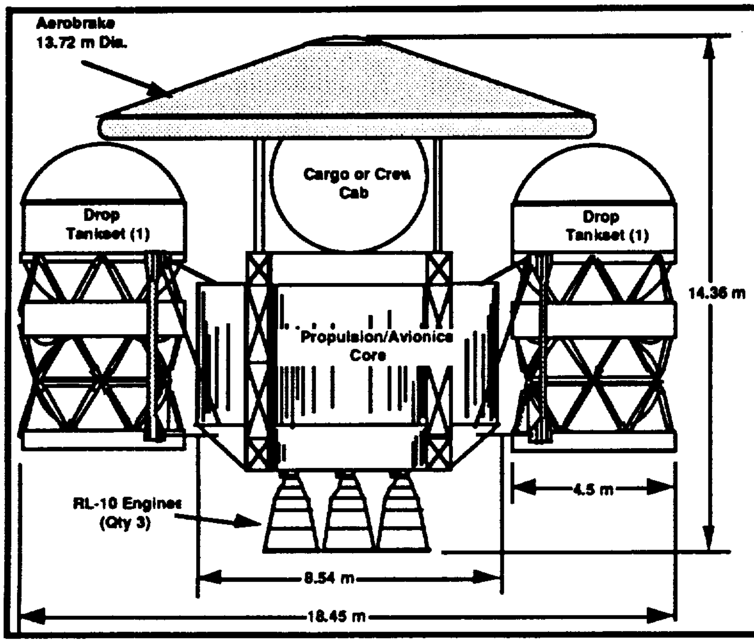
Figure 3.7.1-1 Ground-Based Expendable Version

Table 3.7.1.1 Mass Properties Breakdown - Initial GB STV Configuration

	DESCRIPTION	MASS	MASS	MASS
	GROUND BASE VEHICLE SUMMARY	KG	KG	M.TONS
2	STRUCTURE		681.68	0.68
3	PROPELLANT TANKS		521.77	0.52
4	PROPULSION SYSTEM		309.00	0.31
5	MAIN ENGINES		310.20	0.31
6	RCS SYSTEM		85.03	0.09
7	G. N. & C.		73.92	0.07
8	COMMUNICATION & DATA HNDLG		147.39	0.15
9	ELECTRICAL POWER		250.11	0.25
10	THERMAL CONTROL SYSTEM		377.32	0.38
19	GROWTH		413.46	0.41
	DRY WEIGHT		3169.88	3.17
15	PROPELLANTS		27891.02	27.89
1	LH2	23842.86		
2	LO2	3973.81		
4	FLUIDS & PRESSURANTS	74.35		
18	CARGO		0.00	0.00
	TOTAL WEIGHT		31060.90	31.06

3.7.2 Reusable Initial Concept

This STV, a space-based reusable version (Figure 3.7.2-1), can also be built from the common set of elements and subsystems. Two common tanksets, three engines, an aerobrake, and a core vehicle with limited subsystems form the basis for this vehicle. The dry weight of the vehicle is about 12 tonnes with an assembled length of over 14 m and width of over 18 m. The extra propellant tanksets provide an enhanced performance capability for delivery and return of geosynchronous payloads. The payload can either be deliverable cargo or for some missions a crew module with crew.



Mass Properties	
Components	Mass (t)
Structure	2.3
Propellant Tanks	0.8
Propulsion System	4.3
Main Engines	1.1
RCS System	0.1
GN&C	0.2
Communication & Data Handling	0.2
Electrical Power	0.4
Thermal Control System	0.5
Aerobrake	1.3
Contingency (15%)	1.5
Total Dry Weight	12.1

Performance:
Satisfies DRM E - 1(4.0 t GEO Delivery & Return) and DRM E - 4 (3.5 t GEO Delivery & Return) with
2 Partially Filled Drop Tanksets

Figure 3.7.2-1 Space-Based Reusable Version

3.7.2.1 Mass Properties

Table 3.7.2.1 gives the top level mass properties breakdown for the space-based reusable STV.

Table 3.7.2.1 Mass Properties Breakdown - SB STV Configuration

	DESCRIPTION	MASS	MASS	MASS
	SB STV SUMMARY	KG	KG	M.TONS
2	STRUCTURE		2360.00	2.36
3	PROPELLANT TANKS		802.86	0.80
4	PROPULSION SYSTEM		380.34	0.38
5	MAIN ENGINES		690.00	0.69
6	RCS SYSTEM		122.45	0.12
7	G. N. & C.		195.46	0.20
8	COMMUNICATION & DATA HNDLG		242.70	0.24
9	ELECTRICAL POWER		444.22	0.44
10	THERMAL CONTROL SYSTEM		553.47	0.55
11	AEROBRAKE		1370.00	1.37
19	GROWTH		1074.22	1.07
	DRY WEIGHT		8235.72	8.24
12	CREW MODULE		0.00	0.00
15	PROPELLANTS		95183.00	95.18
	1 CORE TANKS	32000.00		
	2 DROP TANKS	56000.00		
	3 RETURN TANKS	7100.00		
	4 FLUIDS & PRESSURANTS	83.00		0.00
17	PERSONNEL		0.00	0.00
18	CARGO		0.00	0.00
	TOTAL WEIGHT		103418.72	103.42

4.0 STV OPERATIONS

Based on the defined LTS configuration described in section 3.0, the LTS operations concept that will be addressed in this section identifies the ground processing requirements for preparing elements for launch to LEO, Earth-To-Orbit (ETO) transportation of the configuration elements, assembly & checkout of the system at LEO, flight operations from LEO to LLO, decent and ascent and LLO rendezvous and docking, flight operations from LLO to LEO, and post flight checkout and refurbishment of the system. Figure 4.0-1 shows an overview of the elements required to perform the lunar mission. Other elements of this concept that currently have not be defined include direct injection (ground-based) systems and GEO and polar flight operations.

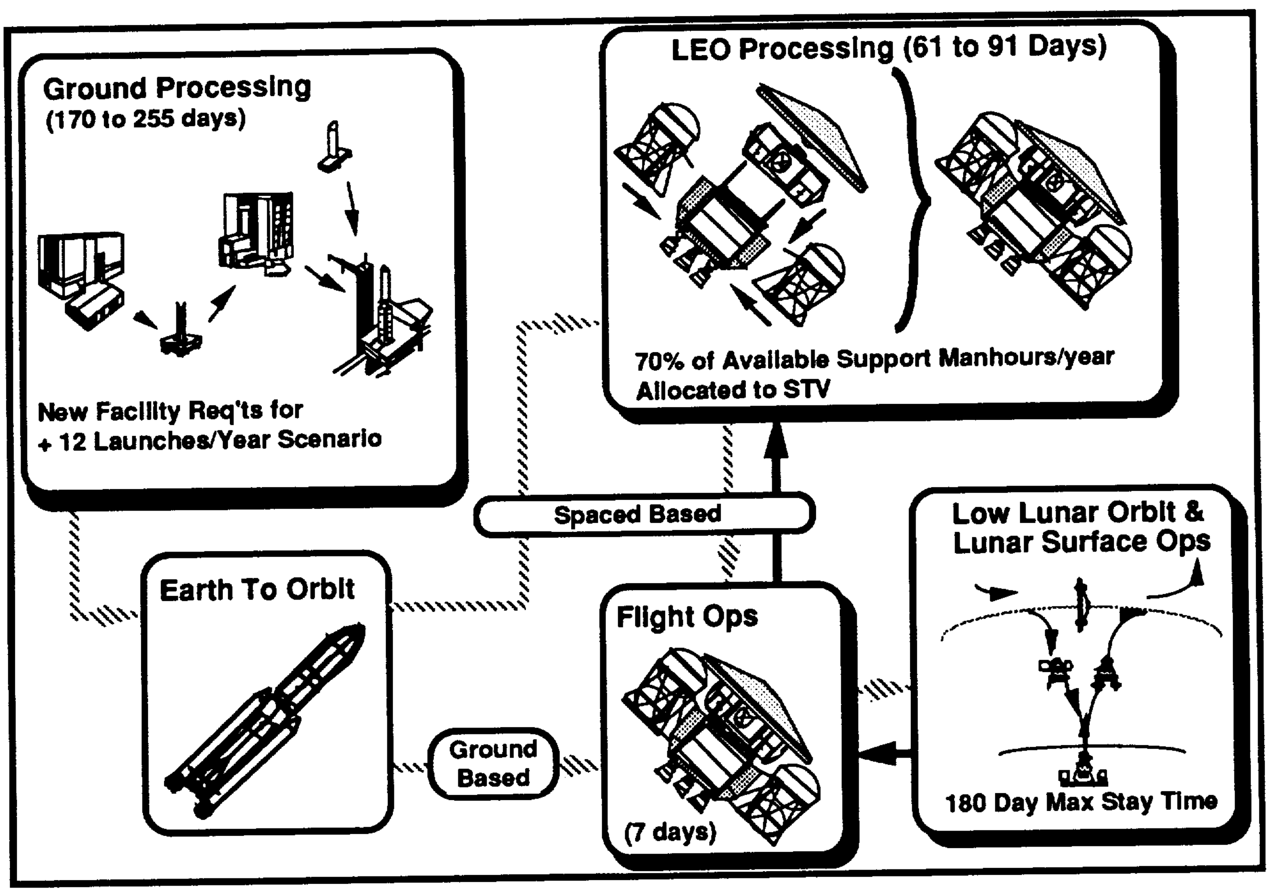


Figure 4.0-1: STV Operations Scenario

This scenario is designed to support the current "Option 5" mission as defined in the Space Exploration Initiative (SEI) plan and supplement in the STV DRM requirements. Tables 4.0-1 and 4.0-2 provide the manifesting plan to support both the lunar and near-Earth missions, which are the baseline for the details defined by the STV operations scenario.

Date	STV Flight	DRM Flight	Mission Type	Transportation Requirement	Mission Configuration	Vehicle Number
2001	1	Near Earth - 1	Cargo	6.4	NE - Expended	NETS -1
2001	2	Near Earth -2	Cargo	3.5	NE - Delivery (1)	NETS - 2
2001	3	Planetary - 1	Cargo	16.0	Expended	PTS-1
01/2002	**4**	**Lunar - 0**	**Cargo**	**33.0**	**Lunar - Expended**	**LTS-1**
07/2002	5	Near Earth - 3	Cargo	6.4	NE - Expended	NETS - 3
2003	6	Near Earth - 4	Cargo	6.4	NE - Expended	NETS - 4
2003	7	Near Earth - 5	Cargo	3.5	NE - Reuse (2)	-
07/2003	**8**	**Lunar - 1**	**Cargo**	**33.0**	**Lunar - Delivery (1)**	**LTS-2**
03/2004	**9**	**Lunar - 2**	**Piloted**	**14.6**	**Lunar - Reuse (2)**	**-**
07/2004	10	Near Earth - 6	Cargo	10.0	NE - Expended	NETS - 5
01/2005	**11**	**Lunar - 3**	**Piloted**	**14.6**	**Lunar - Reuse (3)**	**-**
03/2005	12	Near Earth - 7	Cargo	6.4	NE - Expended	NETS - 6
07/2005	13	Near Earth - 8	Cargo	6.4	NE - Expended	NETS - 7
01/2006	**14**	**Lunar - 4**	**Cargo**	**33.0**	**Lunar - Reuse (4)**	**-**
03/2006	15	Near Earth - 9	Cargo	3.5	NE - Reuse (3)	-
07/2006	16	Near Earth - 10	Piloted	4.0	NE - Reuse (4)	-
01/2007	17	Near Earth - 11	Cargo	3.5	NE - Reuse (5)	-
03/2007	18	Near Earth - 12	Piloted	4.0	NE - Replacement (1)	NETS - 8
07/2007	**19**	**Lunar - 5**	**Piloted**	**14.6**	**Lunar - Replacement (1)**	**LTS - 3**
01/2008	20	Near Earth - 13	Cargo	6.4	NE - Expended	NETS - 9
03/2008	**21**	**Lunar - 6**	**Piloted**	**14.6**	**Lunar - Reuse (2)**	**-**

Table 4.0-1: Lunar and Near-Earth Mission Manifest

Date	STV Flight	DRM Flight	Mission Type	Transportation Requirement	Mission Configuration	Vehicle Number
07/2008	22	Near Earth - 14	Cargo	3.5	NE - Reuse (2)	-
01/2009	**23**	**Lunar - 7**	**Piloted**	**14.6**	**Lunar - Reuse (3)**	**-**
07/2009	24	Near Earth - 15	Cargo	6.4	NE - Expended	NETS - 10
01/2010	**25**	**Lunar - 8**	**Cargo**	**33.0**	**Lunar - Reuse (4)**	**-**
07/2011	**26**	**Lunar - 9**	**Piloted**	**14.6**	**Lunar - Expended (5)**	**-**
01/2012	27	Near Earth - 16	Cargo	6.4	NE - Expended	NETS - 11
03/2012	**28**	**Lunar - 10**	**Piloted**	**14.6**	**Lunar - Replacement (1)**	**LTS - 4**
01/2013	29	Near Earth - 17	Cargo	6.4	NE - Expended	NETS - 12
07/2013	**30**	**Lunar - 11**	**Piloted**	**14.6**	**Lunar - Reuse (2)**	**-**
01/2014	**31**	**Lunar - 12**	**Piloted**	**14.6**	**Lunar - Reuse (3)**	**-**
03/2014	32	Near Earth - 18	Cargo	6.4	NE - Expended	NETS - 13
01/2015	**33**	**Lunar - 13**	**Piloted**	**14.6**	**Lunar - Reuse (4)**	**-**
01/2016	**34**	**Lunar - 14**	**Piloted**	**14.6**	**Lunar - Expended (5)**	**-**
01/2017	35	Near Earth - 19	Cargo	6.4	NE - Expended	NETS - 14
07/2017	**36**	**Lunar - 15**	**Piloted**	**14.6**	**Lunar - Replacement (1)**	**LTS - 5**
01/2018	37	Near Earth - 20	Cargo	6.4	NE - Expended	NETS - 15
03/2018	**38**	**Lunar - 16**	**Piloted**	**14.6**	**Lunar - Reuse (2)**	**-**
01/2019	**39**	**Lunar - 17**	**Piloted**	**14.6**	**Lunar - Reuse (3)**	**-**
03/2019	40	Near Earth - 21	Cargo	6.4	NE - Expended	NETS - 16
01/2020	**41**	**Lunar - 18**	**Piloted**	**14.6**	**Lunar - Reuse (4)**	**-**
01/2021	**42**	**Lunar - 19**	**Piloted**	**14.6**	**Lunar - Expended (5)**	**-**
07/2022	**43**	**Lunar - 20**	**Piloted**	**14.6**	**Lunar - Replacement (1)**	**LTS - 6**
03/2023	**44**	**Lunar - 21**	**Piloted**	**14.6**	**Lunar - Reuse (2)**	**-**
01/2024	**45**	**Lunar - 22**	**Piloted**	**14.6**	**Lunar - Reuse (3)**	**-**
07/2025	**46**	**Lunar - 23**	**Piloted**	**14.6**	**Lunar - Reuse (4)**	**-**
03/2026	**47**	**Lunar - 24**	**Piloted**	**14.6**	**Lunar - Expended (5)**	**-**

Table 4.0-2: Lunar and Near Earth Mission Manifest

4.1 Ground Operations

The present STS shuttle orbiter undergoes stand-alone refurbishment and preparations in the Orbiter Processing Facility (OPF) where payloads can be horizontally installed. The Orbiter is then towed to the Vehicle Assembly Building (VAB) and mated and connected to the ET/SRB stack on the Mobile Launch Platform (MLP) in an integration cell where the entire stack undergoes interface and integration tests. The STS on the MLP is then moved to the launch pad (LC-39A or B) for servicing, checkout, propellant loading, pre-launch and launch, as shown in Figure 4.1-1. It is this processing flow that is the basis for the development of the LTS/STV ground operations scenario.

The LTS/STV vehicle has a modular configuration and consists of the crew module, core vehicle module, aerobrake module, TLI/LOI/RET Tankset modules, and cargo modules. These modules will be processed individually on the ground, manifested and carried to orbit in the payload shroud of the HLLV, and assembled in orbit at space station.

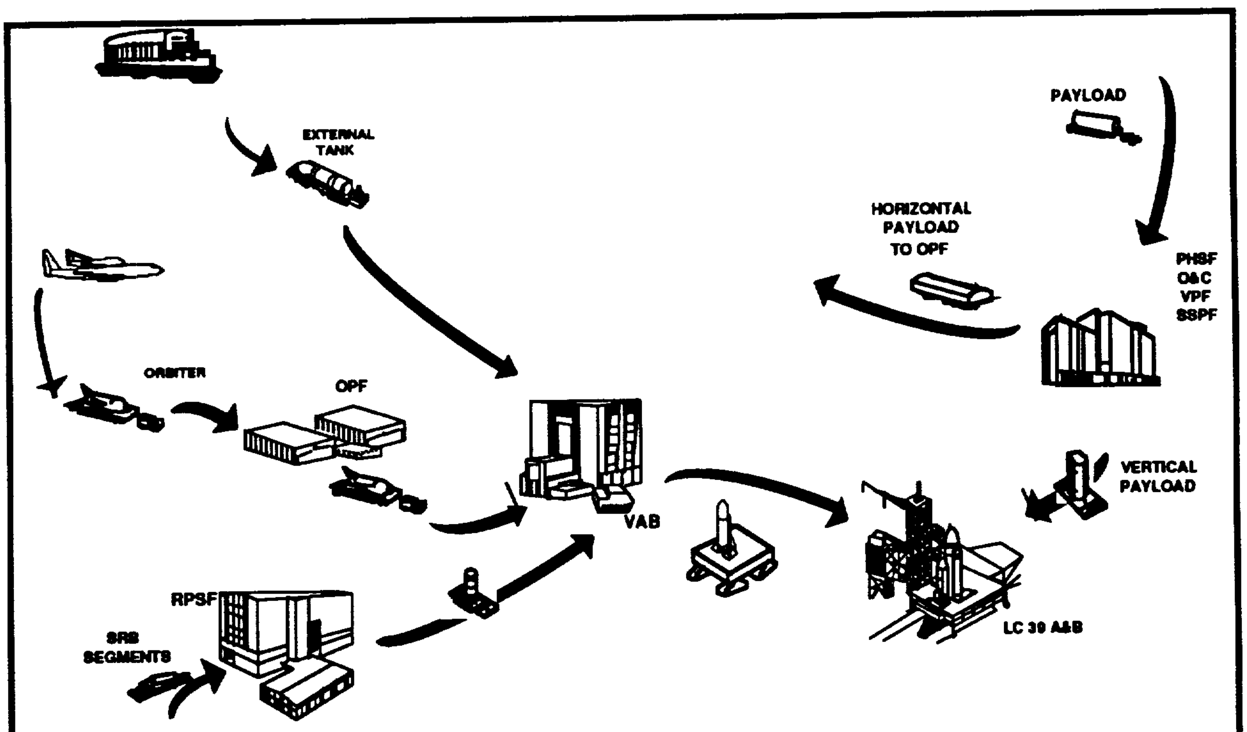


Figure 4.1-1: STS Ground Operations Flow

4.1.1 LTS/STV Ground Operations.

The LTS/STV is considered a payload for the HLLV while simultaneously carrying cargo modules of its own. Stand alone processing for STV modules and vertical integration into the HLLVs payload shroud will be performed in a new combined STV Processing & Integration Facility (SPIF). However, shroud integration and STV module processing could be separate facilities and should be the subject of future study.

Processing of LTS/STV at KSC begins with the receipt of system modules by air and/or barge. These components are then transferred to the SPIF for stand alone processing and subsequent installation into the HLLVs P/L Shroud. The integrated STV/shroud is then transferred to the VAB for mate and integration into the HLLV. After interface testing is complete in the VAB the entire stack is moved to launch pad LC-39C for final HLLV checkout, servicing and launch.

The SPIF consists of the main vertical integration cell and five module preparation cells (Fig. 4.1.1-1), one for each of the LTS/STV sub-modules and cargo, as follows:

Integration Cell	-install STV into HLLV P/L Shroud.
Core/Crew Cell	-receive, C/O & prep. the LLV.
Aero Brake Cell	-receive, C/O & prep. Aero Brake Module
TLI/LOI/Ret Tank Cell	-receive, C/O & prep. Tank Modules
Cargo Cell	-receive & prep. Cargo Module

The integration cell is a large high bay cell where the prepared modules of the LTS/STV are vertically installed into the payload shroud of the HLLV and undergo interface testing. The LLV cell is a low bay cell where the lunar landing vehicle consisting of the crew module (Fig. 4.1.1-2), core module (Fig. 4.1.1-3) and propulsion system are fully functionally tested and receive final sub-system closeout preparations prior to installation into the HLLV payload shroud. The aerobrake cell is a low bay cell where the aerobrake components are received, assembled into flight configuration for full functional testing and prepared for final sub-system closeout prior to brake-down and installation into the HLLV payload shroud (Fig. 4.1.1-4).

The integration cell is a large high bay cell where the prepared modules of the LTS/STV are vertically installed into the payload shroud of the HLLV and undergo interface testing. The LLV cell is a low bay cell where the lunar landing vehicle consisting of the crew module (Fig.

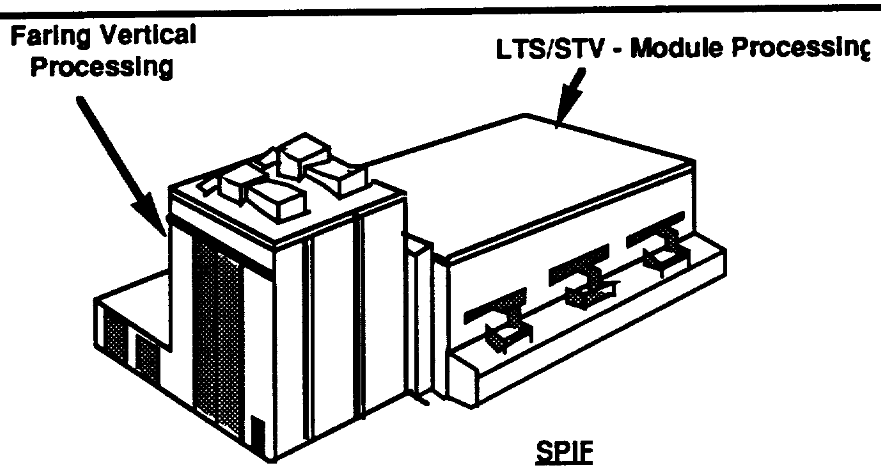


Figure 4.1.1-1: Shroud Processing and Integration Facility (SPIF)

4.1.1-2), core module (Fig. 4.1.1-3) and propulsion system are fully functionally tested and receive final sub-system closeout preparations prior to installation into the HLLV payload shroud. The aerobrake cell is a low bay cell where the aerobrake components (Fig. 4.1.1-4) are received, assembled into flight configuration for full functional testing and final sub-system closeout preparations prior to brake-down and installation into the HLLV payload shroud. The tank module cells are low bay cells where the tank modules (Fig. 4.1.1-5) are received, purged, leak-tested, electrically tested, functionally tested, and final TPS closeout performed. The cargo module cell is a low bay storage cell where cargo modules are temporarily held prior to vertical integration with the LTS/STV and the HLLV payload shroud. The STV Mission Control Center (SMCC), provides 24-hour command and control for the lunar mission and is analogous to the STS Mission Control Center at Johnson Space Center (JSC). Requirements for this center are shown in Table 4.1.1-1.

LTS/STV ground processing takes 50 days of initial stand-alone processing of the basic vehicle with subsequent supporting processing at 20-30 day intervals for tank module flights. The minimum launch interval would be constrained by the launch vehicle and not by LTS/STV. Installation and integration of LTS/STV would occur in the VAB and would not impact any other shuttle processing. Also, loading of the cryogenic propellants could occur the day before launch and have no close-out or impact on the final countdown.

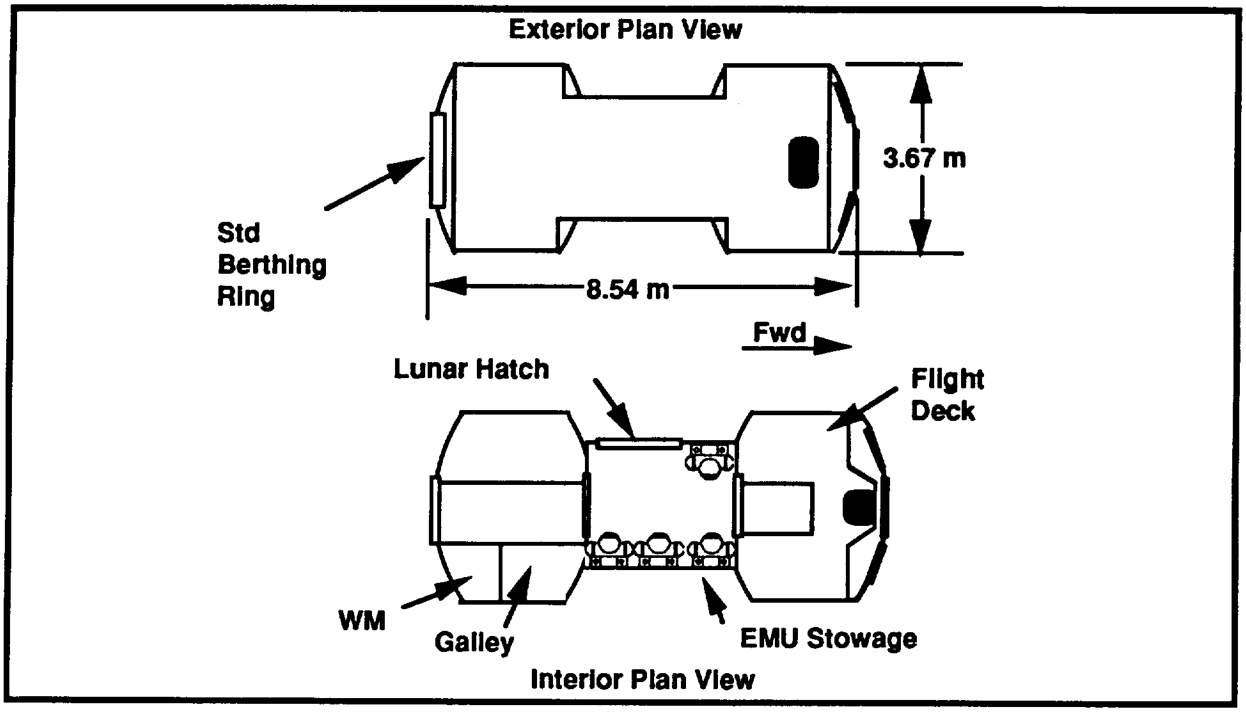


Figure 4.1.1-2: Crew Module

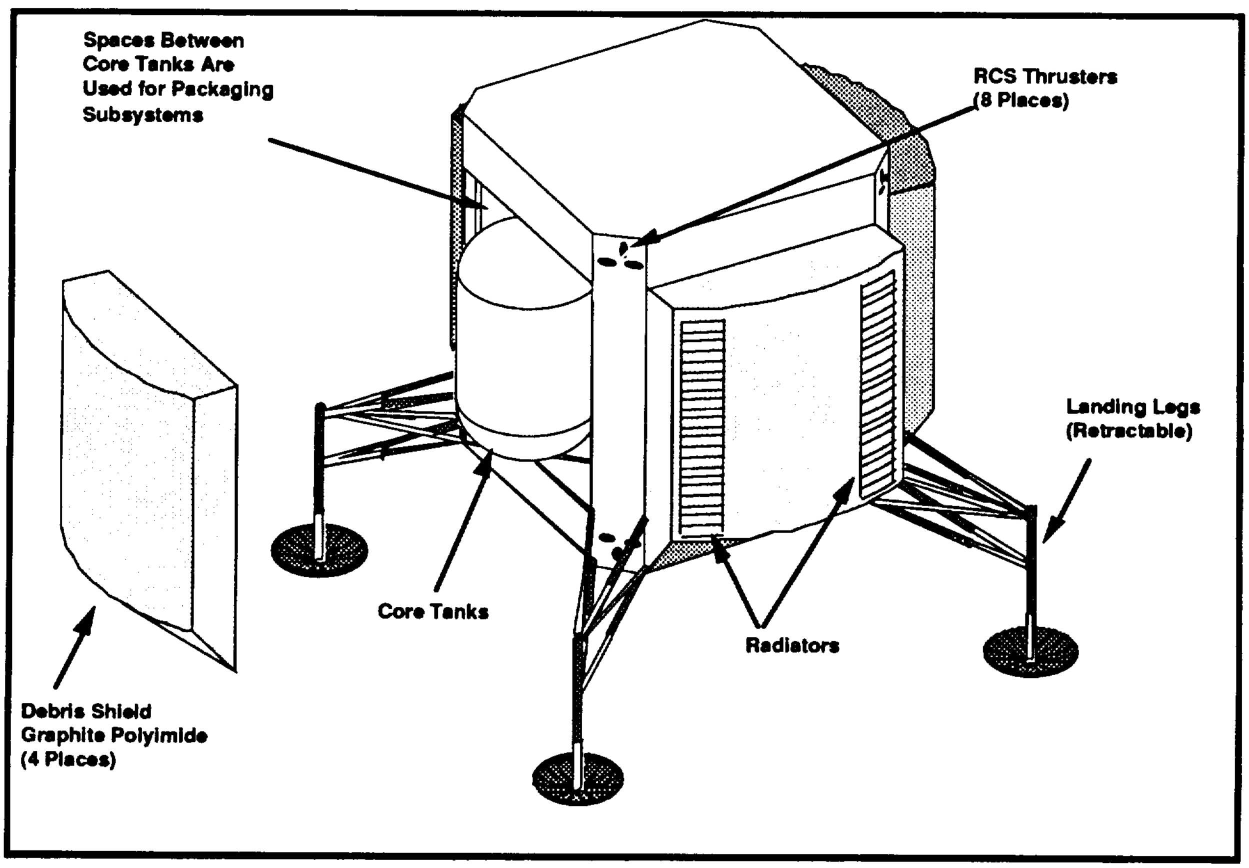


Figure 4.1.1-3: Core Module

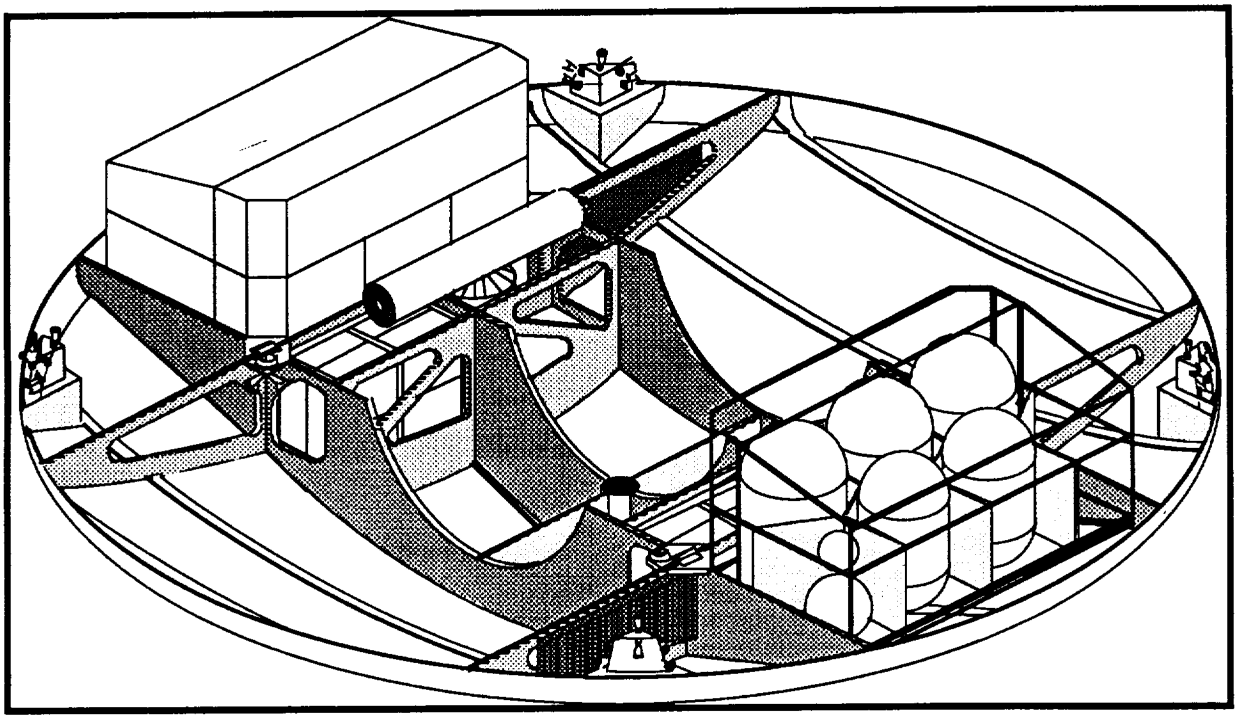


Figure 4.1.1-4: Aerobrake

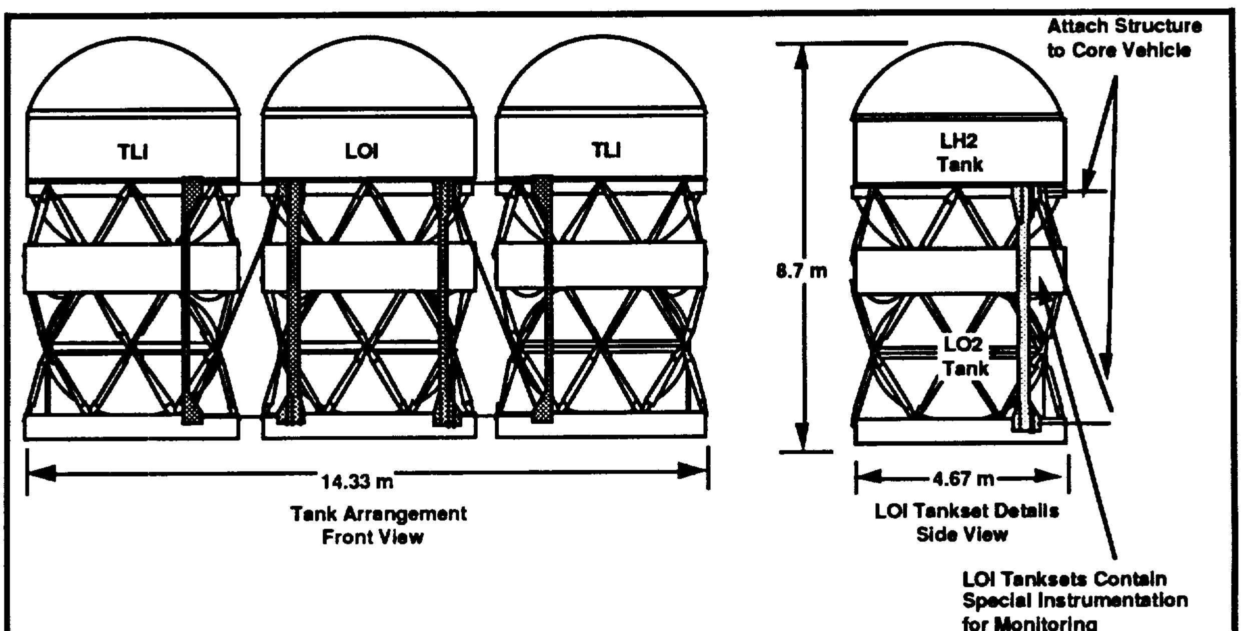


Figure 4.1.1-5: Tank Modules

Table 4.1.1-1: STV Mission Control Center (SMCC)

The Mission Control Center for STV is responsible for the around the clock command and control for all STV flights throughout the entire mission.
The SMCC will be very comparable to the present STS Mission Control Center in all aspects of command, control, communications, data management, facilities, equipment and manpower.
Physical requirements for the SMCC are estimated as follows:
20 to 25 men per shift on a 3/7 week.
Supporting complement of computer terminal work stations.
Supporting computer/network equipment.
Supporting communications network/equipment.
Supporting data management equipment.
Supporting flight/systems simulators.
Supporting software - 9 million lines of code.
Building - 20,000 to 25,000 sq. ft.

4.1.2 ETO Processing and Requirements.

The baseline concept is capable of supporting one lunar mission per year consistent with 'Option-5', - requiring an initial Heavy Lift Launch Vehicle (HLLV) manifest of 3 launches with final STV assembly at SSF. It is planned that STV will be processed and launched at KSC Launch Complex-39 (LC-39) as a payload on a 75 tonnes HLLV ETO launch vehicle. For the purpose of this study it has been assumed that the new HLLV is planned to co-reside with STS shuttle; however, it will have its own dedicated launch pad, LC-39C. Accordingly, processing will be in concert with the existing STS shuttle program and will share integrated processing facilities, support services and range services. Wherever possible, existing facilities are used (Fig. 4.1.2-1). New facilities are identified only where the vehicle design is incompatible with existing facilities or where planned rate usage has saturated facility capacity.

Processing and launch of the LTS elements (Fig. 4.1.2-2) are conducted in six primary tasks and four secondary tasks that involve the processing of the ETO vehicle itself. After receipt, the LTS/STV elements are checked out and integrated into the ETO fairing/shroud, a seventy-five day task. The integrated payload element is then transported to the Vehicle Assembly Building (VAB) for assembly onto the ETO booster element, a ten day task. The completed ETO vehicle is then transferred to the launch pad, where it is processed for launch. The total ground time requirement for the LTS is eighty-five days to launch. To support an initial mission, three ETO flights are required, for a steady state mission, two ETO flights are required. Prior to mating of STV the HLLV is stacked onto the MLP along with its two boosters at the VAB.

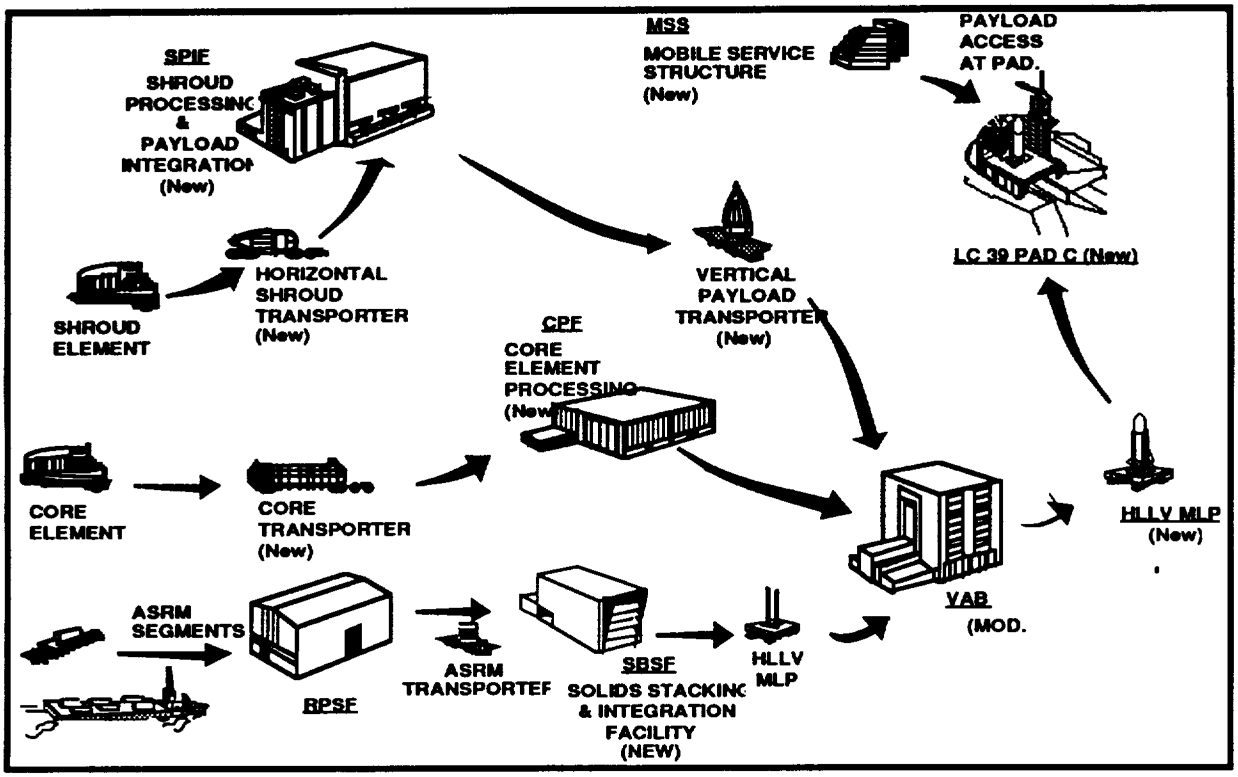


Figure 4.1.2-1: HLLV/ASRM Ground Operations Flow

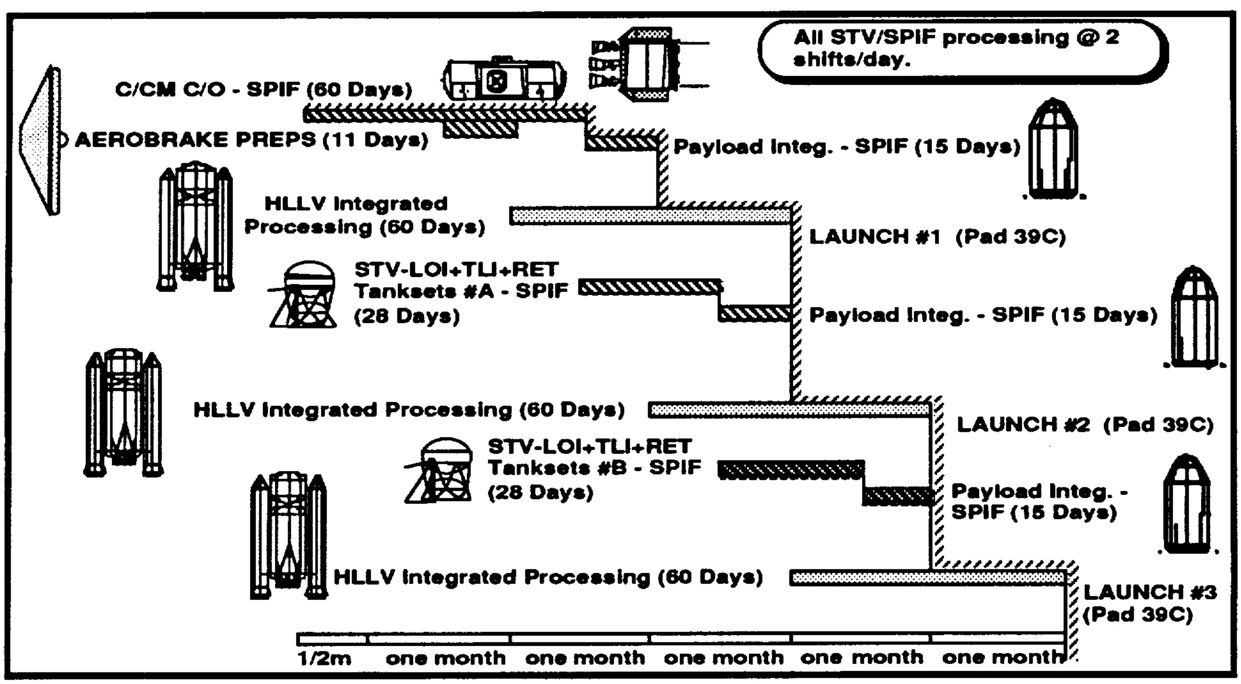


Figure 4.1.2-2 LTS/ETO Processing Flow

The boosters and the HLLV core vehicle have previously been prepared and checked out in their own stand-alone facilities. The Payload Shroud (PLS) containing the LTS/STV is transferred vertically from the SPIF to VAB's transfer isle. The shroud assembly is then hoisted from the transfer isle onto the top of the HLLV stack in the integration cell. Subsequent to the PLS/LTS/STV mate the entire HLLV undergoes interface and integration testing, ordnance is installed and the unit is prepared for roll-out to the launch pad.

Roll out and 'hard-down' takes about 8 hours. After connections to the facility are complete, interface checks are made and a final checkout of the launch vehicle and payload including communications and instrumentation verification is completed. Final servicing (fluids, power, etc.) of all systems is performed just prior to start of the launch countdown. During the launch countdown after all systems power-up, final confidence checks are performed on critical systems and liquid propellants are loaded. LTS/STV propellants will be loaded first and the HLLV is last. After propellants are loaded they will be continuously monitored and vented through pad facilities; at launch the LTS/STV will be locked up and no venting permitted until after booster burnout above 75,000 feet.

The launch site requirements for facilities and major equipment are shown in Table 4.1.2-1. The modifications and new facilities that are required can evolve from the present STS processing system to an STV/HLLV processing system through a logical implementation plan that will minimize the impact to ongoing STS missions. Major new facilities required include a new Launch Pad-C, a SPIF (Shroud Processing & Integration Facility), the MSS (Mobile Service Structure) to accommodate HLLV payload access and/or installation at the pad, a CPF (HLLV Core Processing Facility), and a MLP (Mobile Launch Platform). Also, a major facility modification required includes converting one of the VAB ET cells into an HLLV integration cell, High Bay #2 or #4.

The LTS/STV ground processing manpower requirements per flow are shown in Figure 4.1.2-3 and the requirements for HLLV processing per flow are shown in Figure 4.1.2-4. These two figures are the result of a ground processing task analysis that was based on operational data from the present STS/orbiter processing analogous experience.

The LTS mission scenario requires one mission per year, resulting in an HLLV manifest of three launches with final STV assembly at SSF. Figure 4.1.2-5 shows the relationship of flight manifesting to ETO launches for both the initial mission and for subsequent steady state missions where there is a 20-30 day minimum time between ETO launches. Following

Table 4.1.2-1: Launch Site Facilities and Equipment

LAUNCH VEHICLE SYSTEM >	SHUTTLE/STS (existing - shared/similar)		HLLV	
FACILITIES				
LAUNCH PAD	-		1 PAD-C	New
VERTICAL ASSEMBLY BLDG.	1 VAB (2 INT+2 C/O)		1 VAB (1 INTEG)	Mod.
MOBILE LAUNCH PLATFORM - HLLV	3 MLPs	(similar)	2 HL MLP	New
ASRM PROCESSING	1 RPSF	(shared)	1 RPSF	
SHROUD & PAYLOAD INTEGRATION	-		1 SPIF	New
MOBILE SERVICE STRUCTURE - PAD	-		1 MSS	New
BOOSTER STACK & INTEGRATION	-		1 SBSF	
CORE HORIZONTAL PROCESSING	-		1 CPF	New
STV MISSION CONTROL CENTER	-		1 SMCC	New
TRANSPORTERS				
SHROUD TRANSFER - HORIZONTAL	-		1	New
SHROUD/PAYLOAD - VERTICAL	-		1	New
CORE TRANSFER - HORIZONTAL	0 ET	(similar)	1	New
ASRM TRANSFER - HORIZONTAL	2	(shared)	1	
ASRM TRANSFER - VERTICAL	2	(shared)	1	

Note: Two (2) new MLPs are required to support HLLV launches less than two months apart.

Id.	HLLV ACTIVITY	Bldg.	DAYS	#Techs	Man hours
#1	CORE/CREW MODULE C/O	SPIF	50	10	8,000
#2	AEROBRAKE SECT-A PREPS	SPIF	5	10	800
#3	AEROBRAKE SECT-B PREPS	SPIF	3	10	480
#4	AEROBRAKE SECT-C PREPS	SPIF	3	10	480
#5	TLI TANKSET-A C/O	SPIF	12	12	2,304
#6	TLI TANKSET-B C/O	SPIF	12	12	2,304
#7	LOI TANKSET-A C/O	SPIF	9	12	1,728
#8	LOI TANKSET-B C/O	SPIF	9	12	1,728
#9	RETURN TANKSET-A C/O	SPIF	7	12	1,344
#10	RETURN TANKSET-B C/O	SPIF	7	12	1,344
				TOTAL =	20512
#11	STV+HLLV INTEGRATION	SPIF	15		

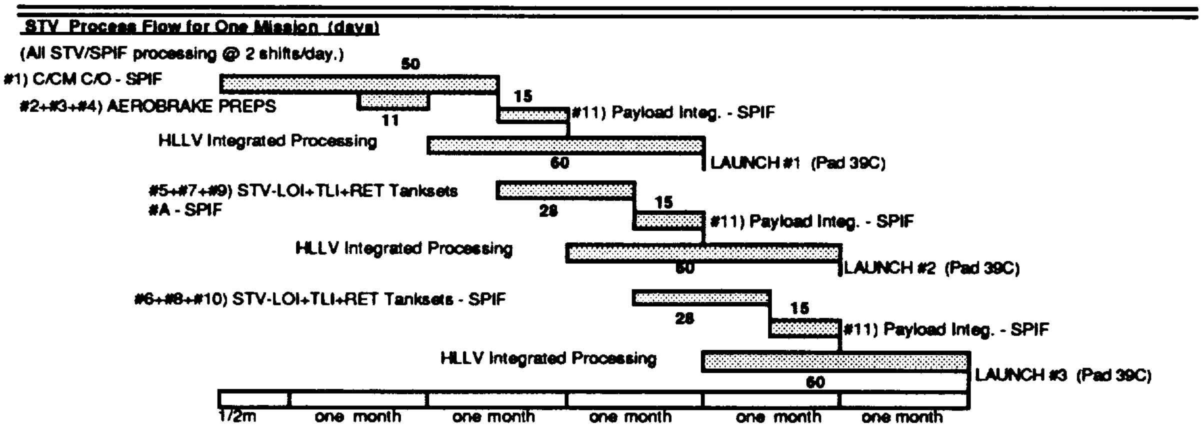
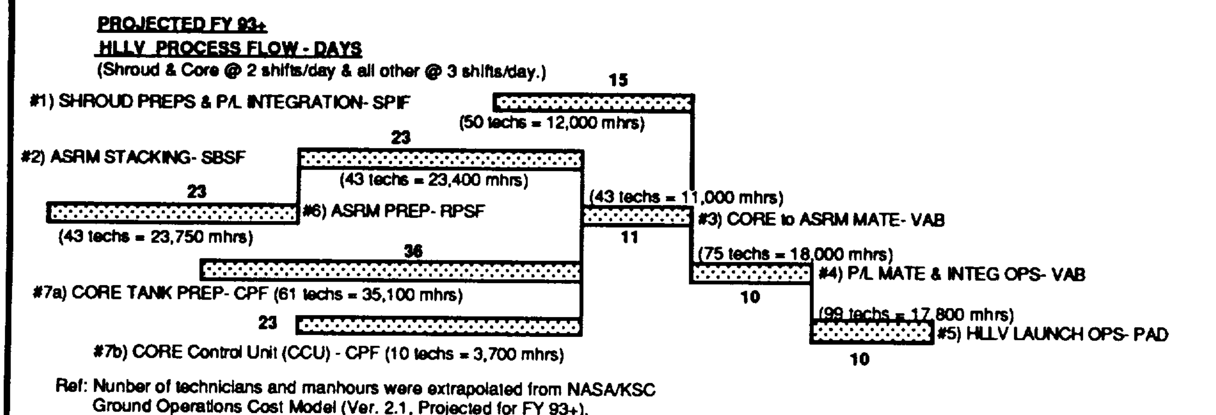


Figure 4.1.2-3: LTS/STV Ground Processing Manpower Requirements

Id.	HLLV ACTIVITY	FACILITY LOCATION	PROJECTED 93+ DAYS
#1	SHROUD-P/L INTEG	SPIF	15
#2	ASRM STACKING	SBSF	23
#3	CORE to ASRM MATE	VAB	11
#4	P/L MATE & INTEG OPS	VAB	10
#5	HLLV OPS	PAD	10
#6	ASRM ELEMENT PREPS	RPSF	23
#7a	CORE ELEMENT PREPS	CPF	36
#7b	CORE CONTROL UNIT C/O	CPF	23
	POST LAUNCH REFURB	PAD/MLP	7
	MLP PATH TOTAL =		61
LAUNCH RATE (MLP limited)/year=		LC-39C	6.0

STAFFING RATIOS	
TECHNICIANS	1.00
PROCESS ENGINEERS	0.89
OVERHEAD/ADMIN	0.42
PLANNING & CONTROL	0.22
FACILITY & GSE	1.85
LOGISTICS	0.53
SR & QA	0.46
TOTAL =	5.37



Ref: Nunber of technicians and manhours were extrapolated from NASA/KSC Ground Operations Cost Model (Ver. 2.1, Projected for FY 93+).

Figure 4.1.2-4: HLLV Processing Manpower Requirements

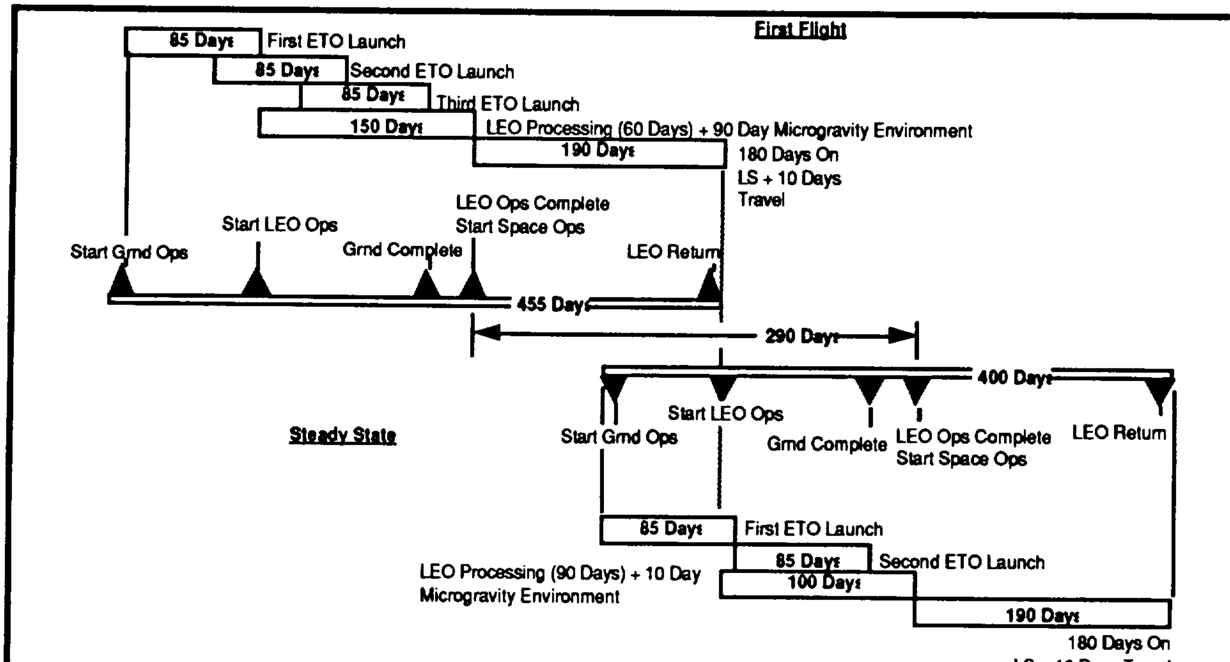


Figure 4.1.2-5: Lunar Mission/Ground Processing Timeline Relationship

comparable STS timelines, processing of HLLV will probably take 65-70 days from receipt of the HLLV core element. The period of close coordination with STS will only be about the 21 days in the VAB and, of course, for launch control the 10 days at Pad-C.

4.2 SPACE OPERATIONS

The space operations for the LTS/STV consists primarily of two phases. The first involves the activities that take place in Low Earth Orbit (LEO) followed secondly by the inflight operations that support the transport of the vehicle from LEO to its destination. In the case of manned missions, the system is returned to LEO for refurbishment and preparation for the next mission. Activities at LEO will be supported, coordinated, and controlled through the SSF command and control systems. The data reported in this section will define the functional sequencing and timelines for the LEO operations including assembly and checkout of the LTS/STV, the inflight operations supporting of the lunar missions, and the post-flight and system refurbishment activities in LEO at the conclusion of the mission.

4.2.1 Low Earth Orbit Operations

The LEO node has been identified as the transportation node for the lunar exploration missions. The primary element of the LEO will be Space Station Freedom (SSF) and its proximity operations support equipment. A general overview of the defined operations in LEO begin with the ETO system delivering LTS hardware elements to an SSF parking location. This point in LEO has been defined as being approximately 20 miles from SSF. Elements of SSF Proximity Operations SE transport these elements back to SSF, where they are received and readied for assembly and checkout. Following the completion of the assembly activity, the system undergoes a final flight readiness verification test. The system is then transfer from SSF to its TLI station again using SSF Proximity Operations SE.

As described above, the LEO operation is initiated with the delivery of LTS hardware elements by the ETO launch system. To complete assembly and launch the LTS, three ETO flights are required for a first flight mission, and two flights are required for the steady state missions. First flight missions are defined as missions that require delivery of an LTS vehicle (consisting of a core module, crew module or cargo platform, and the aerobrake) as well as the necessary propellant quantities. A steady state mission requires only delivery of the propellant since the vehicle elements of the system are being reused. Figure 4.2.1-1 represents the elements delivered by each of the ETO launch as processed on the ground. As described in Section 4.1, launch of the ETO

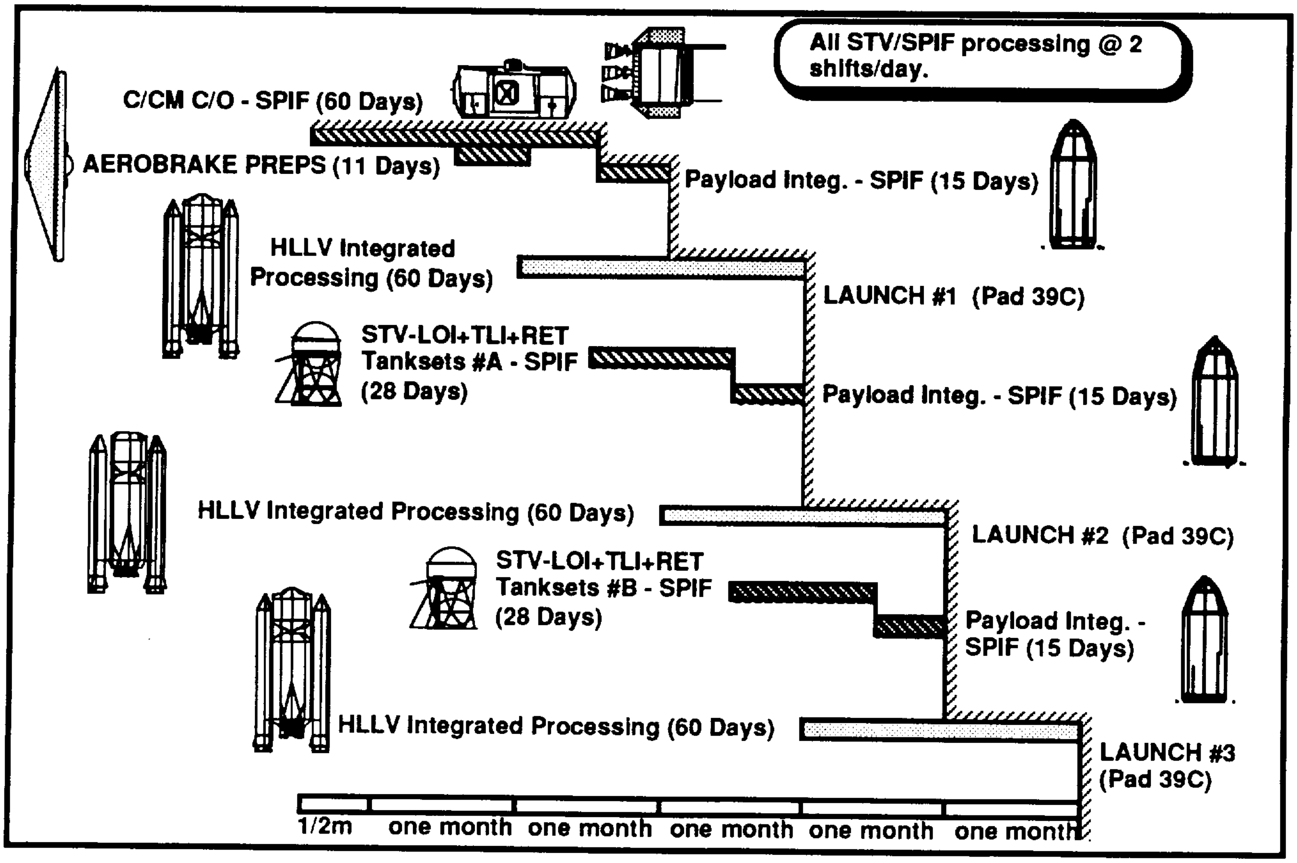


Figure 4.2.1-1: LTS/ETO Payload Allocation

system is integrated with either a payload made up of the core module, the crew module, and the aerobrake, or a three propellant tankset, a process that takes a total of 6-1/2 months for the first flight and 3-1/2 months for the steady state flights.

Figure 4.2.1-1 defines the complete set of timelines for the processing of LTS elements for both the first flight and steady state scenarios. For the initial flight mission, there are six primary activities performed at LEO (SSF). The hardware delivery phase (16.5 days) receives the LTS components at SSF where an element level checkout is conducted. The assembly phase (17.5 days) assembles the LTS components into an operational configuration. This is followed by the verification phase (16 days) that ensures flight readiness of the system. With the system mission ready, the propellant servicing phase (9 days) assembles the drop tanks to the mission vehicle. The closeout phase (9 days) provides final launch readiness, and is followed by the launch phase (2.5 days). The launch phase delivers the mission crew, transports the LTS to the injection burn location, and initiates TLI. Total processing time for an initial flight mission is 61 days, although due to the KSC launch window constraints of 30 days, the actual time required to process the LTS is 265 days. Figure 4.2.1-3 breaks down the processing functions for the first flight tasks to show

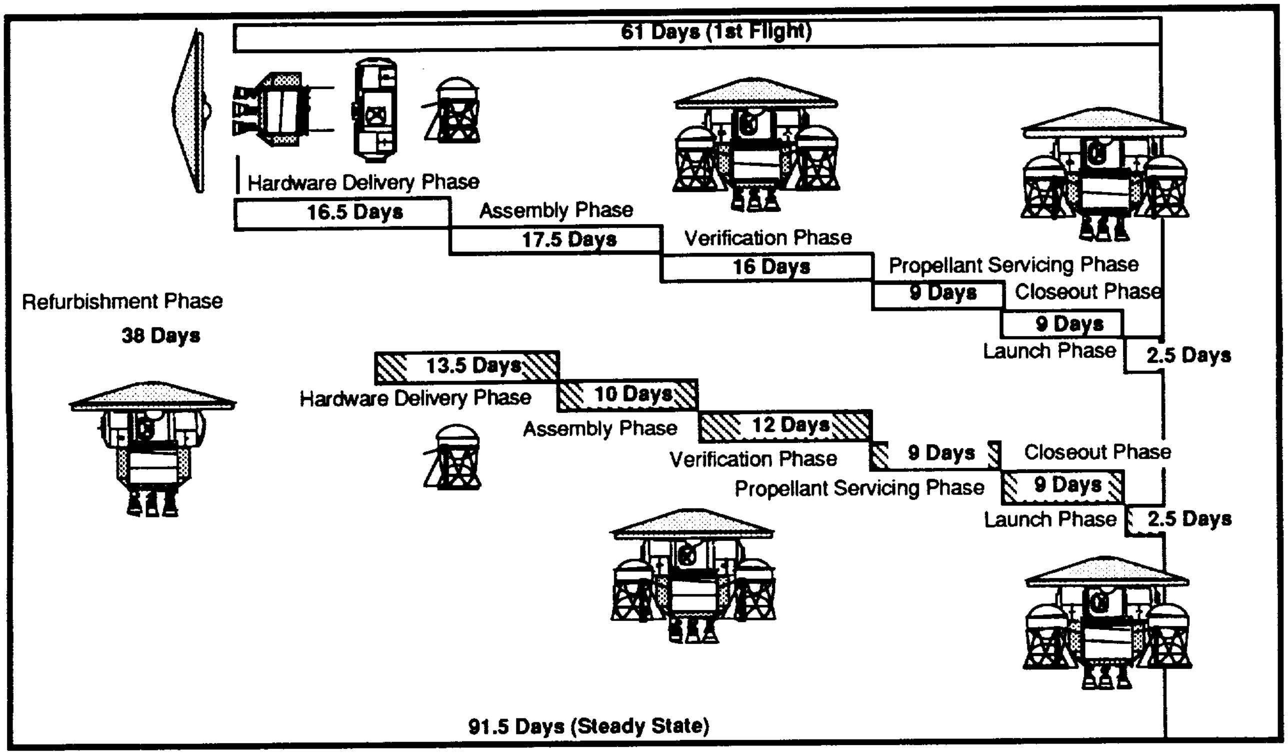


Figure 4.2.1-2: LTS Processing Timelines

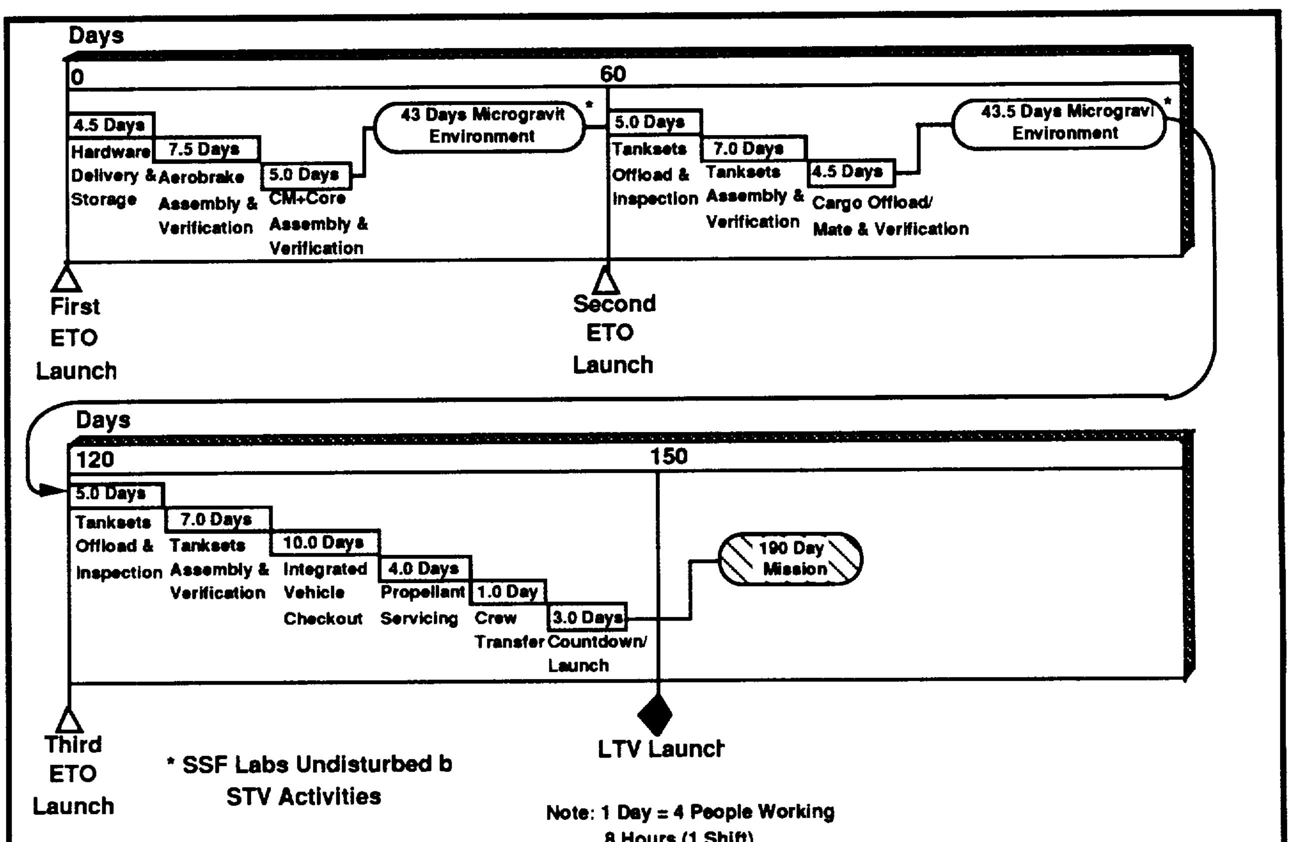


Figure 4.2.1-3: Detailed Processing Timelines For the Initial Flight Scenario

the specific function sequences and timelines and provides insight into where and how the processing downtime due to the KSC launch windows affect the task. These downtimes represent a total of 86 of the 265 processing days, where there the microgravity environment of SSF is undisturbed.

For the steady state missions the number of processing steps increases from six to seven, although the time required for many of the phases is reduced based on the vehicle configuration that returns to LEO following that mission. The first processing phase of a steady state mission is the refurbishment phase (38 days), where the returning LTS is completely checked out and refurbished. The hardware delivery phase (13.5 days) receives the propellant tanksets at SSF where an element level checkout is conducted. The assembly phase (10 days) assembles the replaceable LTS components into an operational configuration. This is followed by the verification phase (12 days) that ensures flight readiness of the system. With the system mission ready, the propellant servicing phase (9 days) assembles the drop tanks to the mission vehicle. The closeout phase (9 days) provides final launch readiness, and is followed by the launch phase (2.5 days) which delivers the mission crew, transports the LTS to the injection burn location, and initiates TLI. Total processing time for an initial flight mission is 91.5 days. Figure 4.2.1-4 breaks down the processing functions for the steady state tasks to show the specific function sequences and timelines. It also provides insight into where and how the previous mission and processing downtime due to KSC launch window constraints affect the overall tasks. These downtimes represent a total of 195 of the 290 available processing days where the microgravity environment of SSF is undisturbed. An estimate of the vehicle refurbishment hours is shown in Figure 4.2.1-5. The vehicle refurbishment hours account for inspection, repair and/or replacement and functionality testing of the major STV components. Some tasks will require both EVA and IVA activity, based on the complexity of the refurbishment. For example, TPS repair on the aerobrake is a delicate operation that can best be completed by an astronaut. The basic inspection of the TPS to identify repair locations can be accomplished through IVA using a camera, FTS and sensors. EVA requires two astronauts to be working externally and one astronaut to be monitoring the activity internally. The monitoring astronaut does not need to be monitoring the entire sequence; therefore, EVA hours account for two astronauts EVA plus one half astronaut IVA. Certain EVA and IVA tasks can be accomplished simultaneously allowing the overall refurbishment to be completed in 76 days. Further refinement on the timeline will be accomplished in future studies. The refurbishment activities will be evaluated considering ground processing, robotics, IVA and EVA. Ground operations in combination with robotics is the more desirable refurbishment option as it minimizes astronaut involvement. Ground operations can be performed 24 hours a day, 7 days a week. The concern of ground uplink delay (~ 3 sec) can be

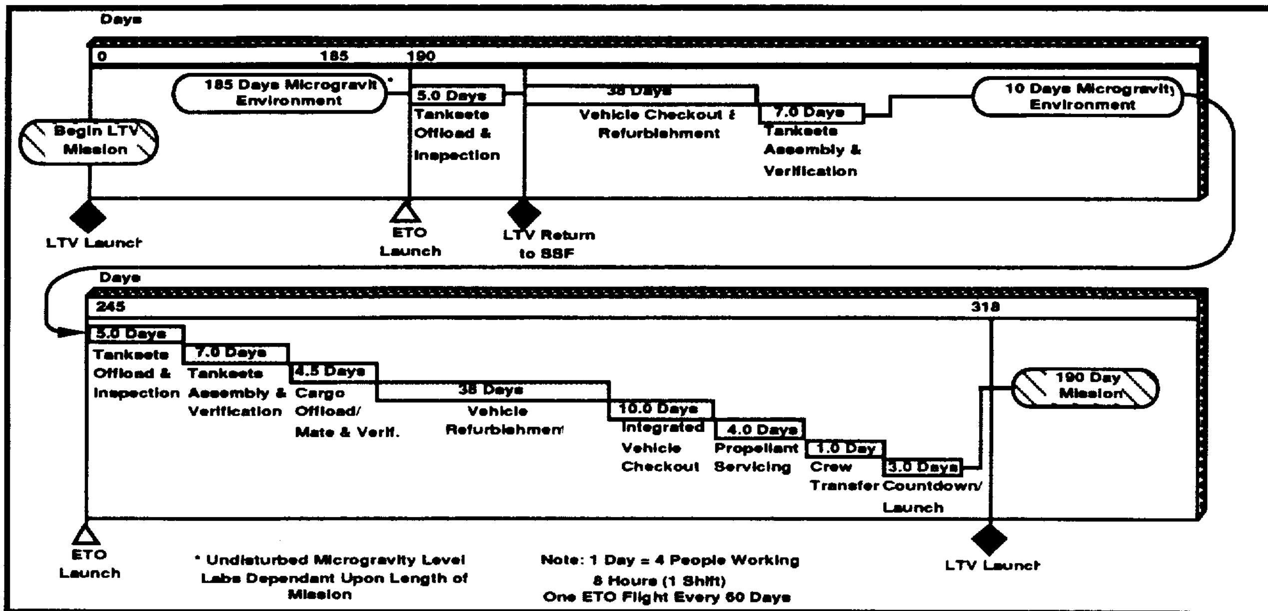


Figure 4.2.1-4: Detailed Processing Timelines - Steady State Flight Scenario

Task	IVA Hours	EVA Hours
Aerobrake		
Inspection	40	24
TPS Repair	40	60
Replacements	as needed	
Functionality Test	20	
Total	100	80
Propulsion System		
Engine Inspection	96	48
Propellant System	64	40
Replacements	as needed	
Functionality Test	24	
Total	184	88
Core		
Avionic/C&T	60	24
Thermal	80	40
Power	80	40
Total	220	104
Crew Cab		
Inspection	40	24
Clean Up	40	
ECLSS	80	24
Avionics	20	
Waste Mgmt	20	
Functionality Test	56	24
Total	256	72
Vehicle Servicing	760	348

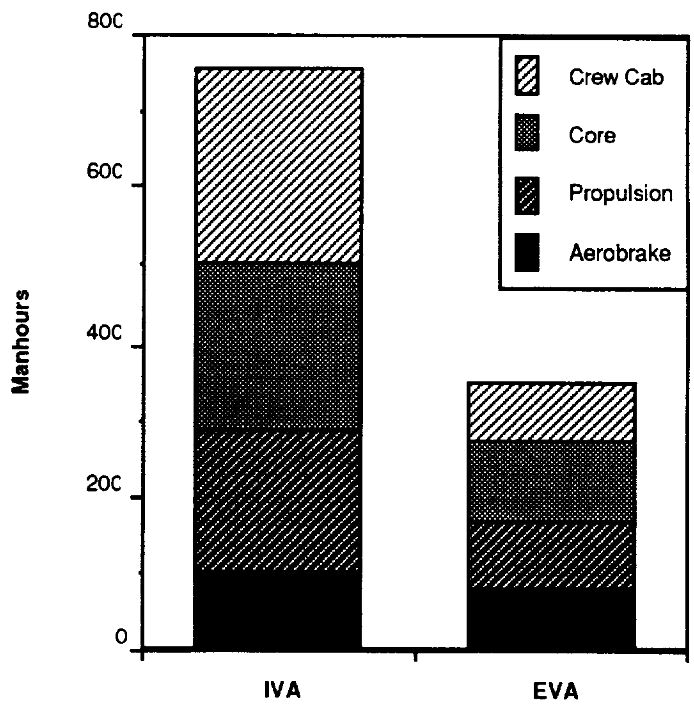


Figure 4.2.1-5: Vehicle Refurbishment Hours

accommodated easily. Current CAD and GEOMOD operations can take 3+ seconds to complete and this time delay can be factored into the operation. Any ground operation should have astronaut override in the event of an emergency. EVA and IVA refurbishment should be used only for delicate operations, intelligent operations, and contingencies.

4.2.2 Space Flight Operations

Once the processing activities at the LEO node have been completed and the LTS transferred away from the node to a remote location, the initial phase of the space flight activates begin. Space flight operations encompass those functions that make up the outbound mission from LEO to low lunar orbit, the rendezvous and docking and station keeping activities in LLO prior to descent and following ascent, descent and ascent to the lunar surface from LLO, and the inbound mission from LLO to LEO and recovery by the LEO node. Figure 4.2.2-1 shows the complete space flight architecture that has been defined for the LTS mission. Although the figure represents a piloted mission, the reusable cargo mission uses the same mission functions and the expendable cargo missions follow the same functions through descent to the lunar surface.

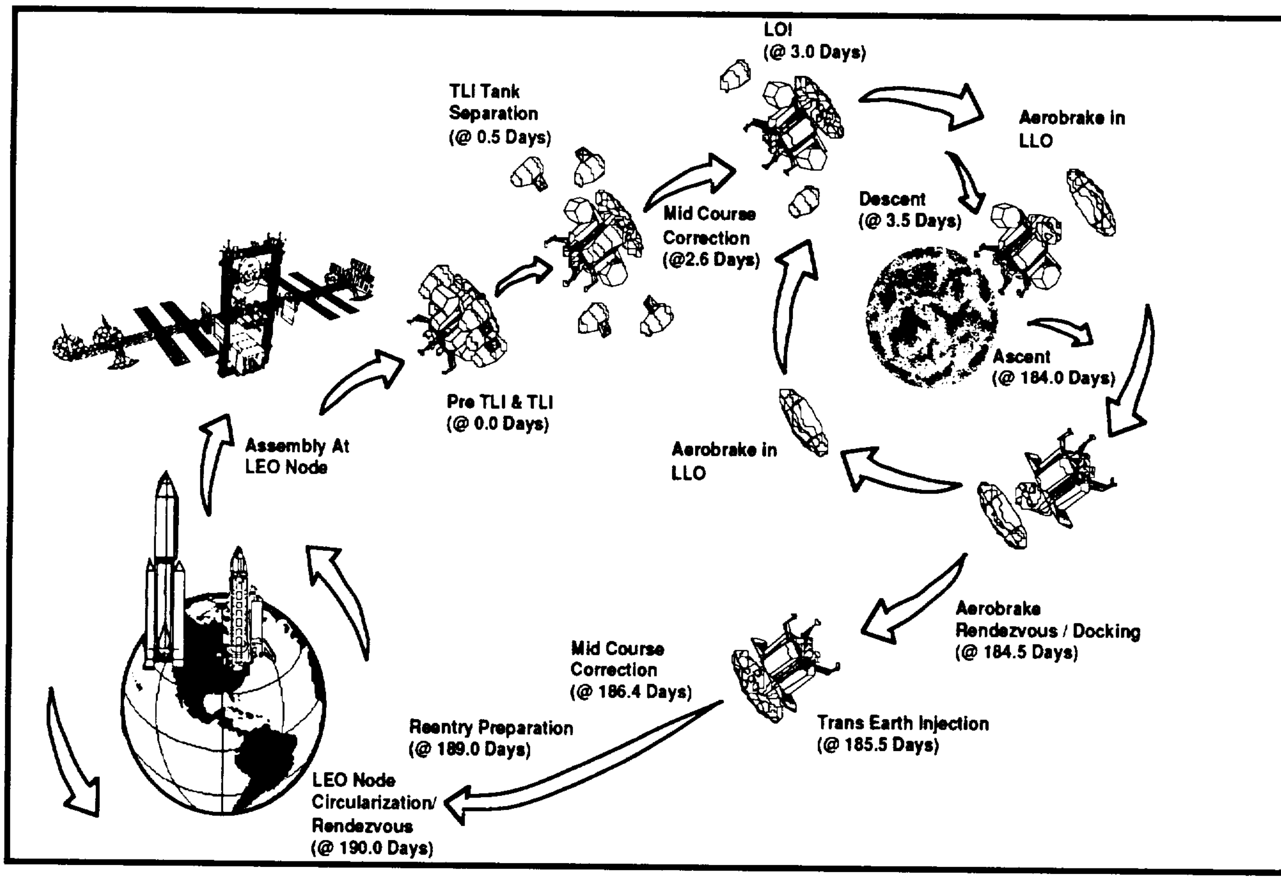


Figure 4.2.2-1: Space Flight Operational Functions and Timelines

The development and selection of this architecture was a primary part of the configuration selection analysis, Section 2.3.3.3. The functions themselves were identified in the early phases of the NASA 90-Day SEI Study and have been refined as the STV Study has matured. Table 4.2.2-1 represents the performance requirements implemented at each of the mission functions. In conjunction, Figure 4.2.2-2 defines the optimum transit time for the outbound and inbound phases

Table 4.2.2-1: LTS Mission Performance Requirements

LEO Ops	10
TLI	3100
Gravity Loss for TLI	150
TCM-1	10
LOI/TEI	1100
Lunar Descent	2000
Lunar Ascent	1900
LLO Operations	50 for TV, 10 for LV
EOC	275
LEO Ops (After EOC)	40
Direct Lunar Descent	3000 *
Direct Lunar Ascent	2900 *

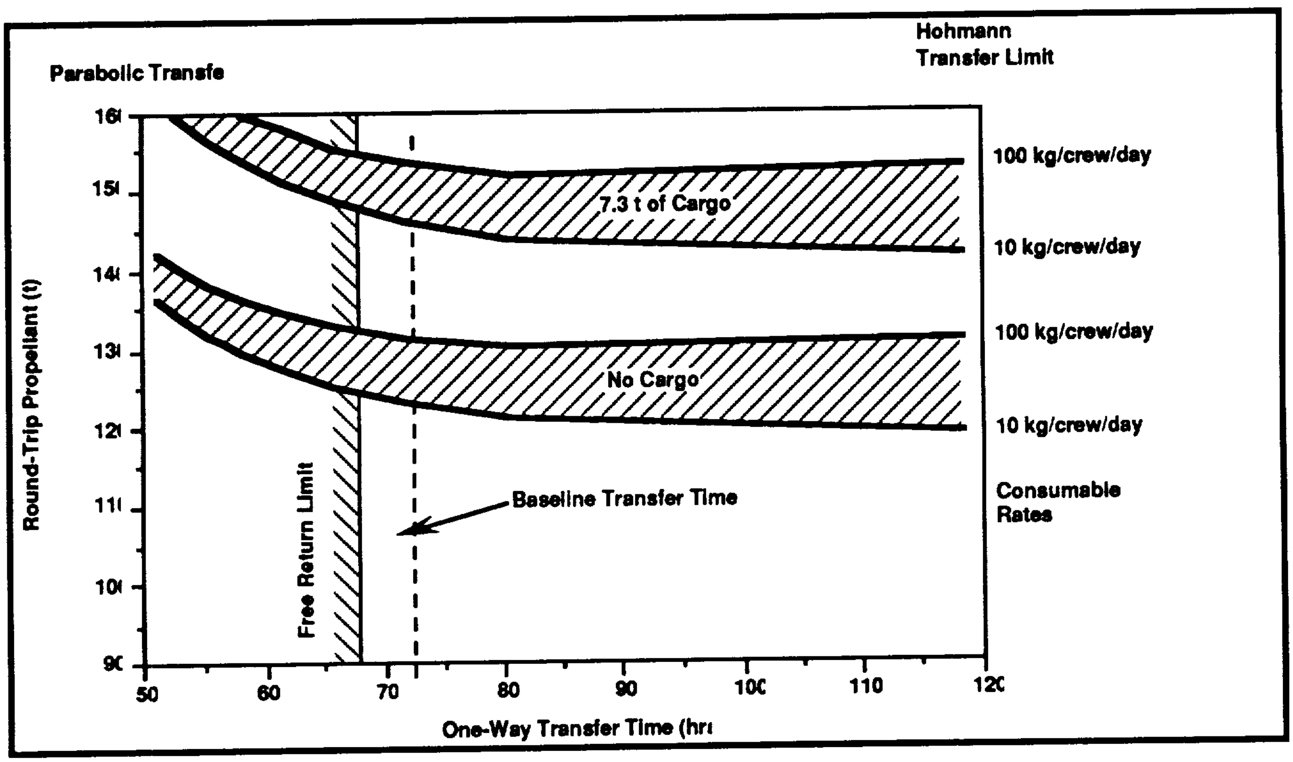


Figure 4.2.2-2: Total Round-Trip Propellant vs Mission Trip

of the mission. The baseline for the lunar stay period was defined by the NASA "Option 5" lunar exploration scenario and has been integrated into the LTS mission scenario.

As noted above, the lunar mission initiates with the LTS in LEO in a pre-TLI mode. This mode allows the flight crew to conduct a series of final flight readiness checks prior to TLI. Once the system is ready for flight, the main engines fire and TLI is performed. This engine burn lasts for 19.4 minutes at which time the LTS transitions from powered flight to a coast mode. One-half a day into the mission (T+0.5 days), the TLI tanks separate from the two tankset assemblies and are placed into an earth escape trajectory where they are destroyed using an internal self-destruction system. A mid-coarse correction burn occurs at T+2.6 days, in preparation for the Lunar Orbit Insertion (LOI) burn at T + 3.0 days. Once the LTS is stabilized in LLO, two function are performed in preparation for descent to the surface. The first function is the separation of the LOI tanks sets and their placement in a controlled trajectory that will impact the lunar surface. This is followed by the separation of the aerobrake stage from the LTS lander stage. Once separated, the aerobrake conducts an independent stabilization maneuver and places itself in a station-keeping mode for the duration of the lunar stay period (1 to 6 months). At T + 3.5 days, the main propulsion system fires and initiates the lunar descent maneuver. The LTS lander leaves LLO descending to the lunar surface and the manned lunar outpost. With the lander on the surface, the payload is unloaded by surface support equipment and, in the case of piloted missions, the crew egresses the lander and transitions to the outpost facilities. These activities are deviated from only on Flight 0, where the payload is the unloading support equipment in which instance it must unload itself. At this time the specifics of the crew handling equipment have not been completely defined, therefore the egress operations could be either IVA or EVA, with the interfaces developed as details become available. After the crew and/or cargo have been removed, the lander system remains powered for an addition forty-eight hours, during which several post flight tests are conducted. Following this period, the primary systems of the lander power down and the propellant is offloaded to a surface storage system, leaving the monitoring systems to be operated from base power for a period of one to six months.

Forty-eight hours prior to ascent, the lander is fueled and all systems powered up. This is followed by a series of pre-flight readiness checks on both the lander and aerobrake. With both systems ready for flight, the lander ascends from the lunar surface to LLO at T+184 days. The launch of the lander is timed to minimize the plane change requirements as well as permitting early rendezvous with the aerobrake. In preparation for rendezvous the lander ascends to a low altitude parking orbit, where a periapsis burn is performed to change the orbital plane and achieve apoapsis behind and below the aerobrake. This is followed by a circularization burn and a series of orbital injection maneuvers and coasts that complete the approach phase. The next step in the rendezvous task is the terminal phase which positions the lander within 100 ft of the aerobrake by matching both the positions and velocities of each vehicle. The two vehicles close on each other along a line-of-sight at very small Δvs with the alignment made either manually or automatically depending on the missions - piloted or cargo. Docking occurs at T + 184.5 days using a "Probe and Drogue" soft docking approach, with the two vehicles hard docked using columns that extend from the lander to corresponding docking mechanisms on the aerobrake. Figure 4.2.2-3 provides an overview of the ascent, rendezvous, and docking activities, and Figure 4.2.2-4 shows the details of the approach, terminal, and docking phases.

Following docking of the aerobrake and lander, the main propulsion system conducts the trans-Earth injection (TEI) burn at T+185.5 days. Propellant for this burn is housed in the aerobrake tanks and is transferred to the engines along feed lines in one of the docking mechanism columns. This engine burn lasts for 62 seconds at which time the LTS transitions from powered flight to a coast mode. A mid-course correction burn occurs at T+186.4 days in preparation for the earth reentry maneuver that is initiated at T+189.0 days. The Earth Orbit Insertion (EOI) burn at T+190 days circularizes the orbit of the LTS to match that of the LEO node. A similar rendezvous and docking activity to that used in LLO is used to close with the LEO node and finally to dock with the LEO Node Proximity Operations support equipment. The mission described above defines the longest duration piloted mission required; shorter missions reduce the total mission time by

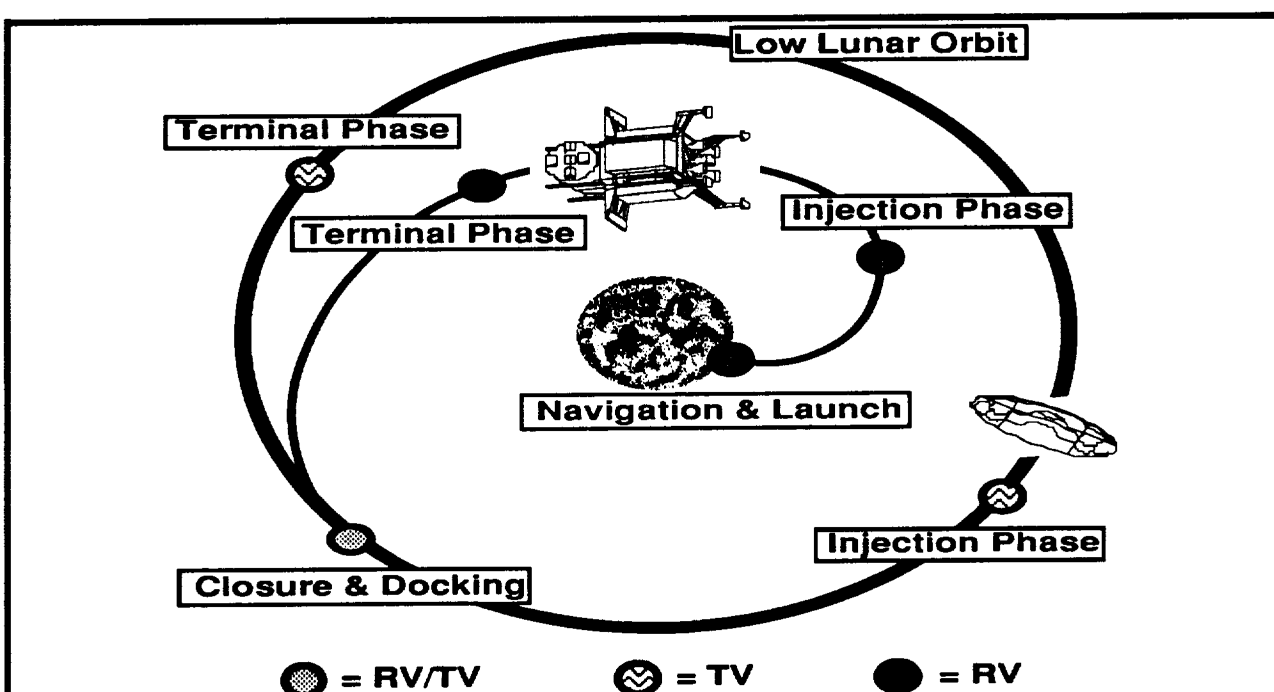


Figure 4.2.2-3: LLO Rendezvous And Docking Overview

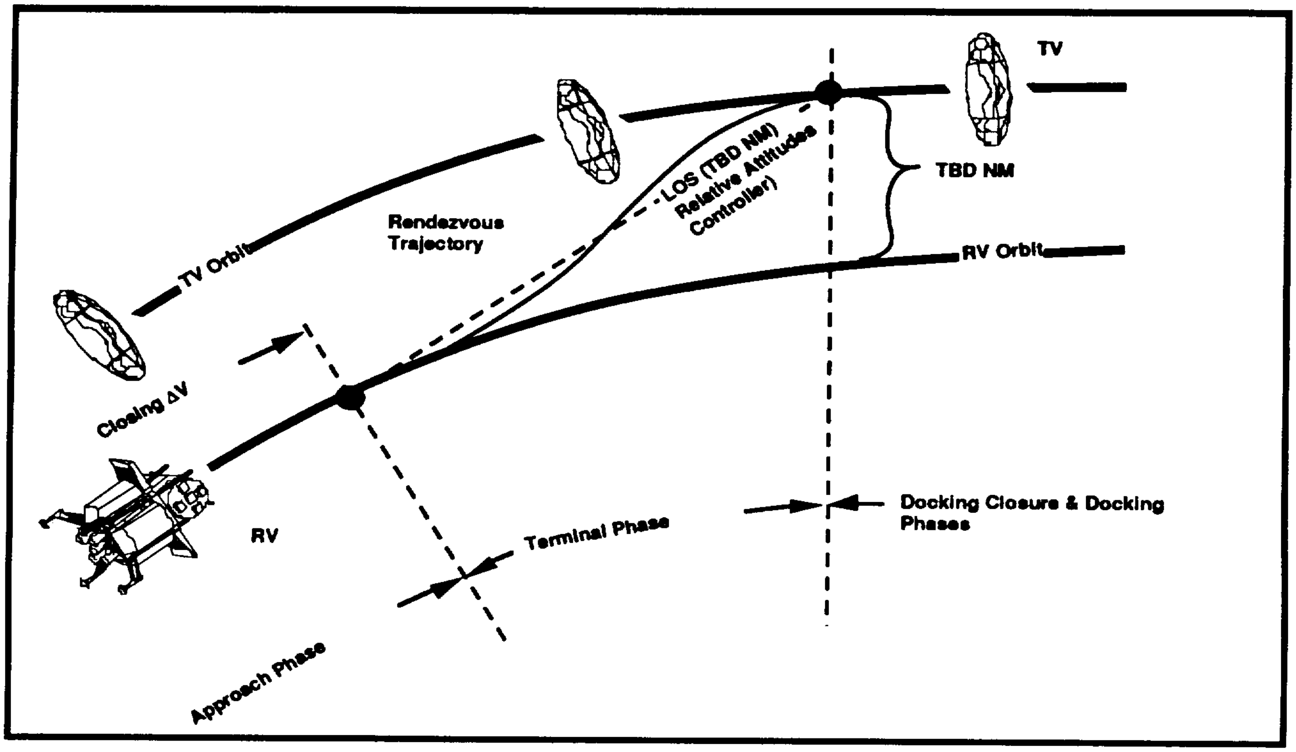


Figure 4.2.2-4: Approach, Terminal, And Docking Phases

shortening the stay time on the surface. Expendable cargo missions are conducted on the outbound and descent activities and employ only minimal surface times. Figure 4.2.2-5 shows the overall mission timeline for a piloted mission, starting with receipt of hardware in LEO, the initial mission, system refurbishment, conduct of a steady state mission including return to the LEO node. Details of the LEO processing phases of this timeline have been defined in section 4.2.1, Ground Processing.

With the functions of the mission defined, each was analyzed for potential failure modes as well as recovery scenarios. Mission rules considering all possible scenarios, failures and recovery methods were generated for each mission. The primary emphasis on any abort is the recovery of the crew, with the primary goal of returning to Earth and secondary goal of placing the vehicle in a position where rescue can be accomplished. In the case of an abort all elements (cargo, propellants, etc.) are considered expendable if the release of these elements increases the possibility of a successful abort and/or rescue. In-flight EVA will be included in these aborts to manually perform those tasks which have not automatically been corrected. Three detailed scenarios have been developed to define generic abort possibilities. The first two sections contain mission phase determined abort scenarios. These are aborts that may occur at a particular time or phase of the mission. The third section is a listing of systems and impacts due to loss of these systems.

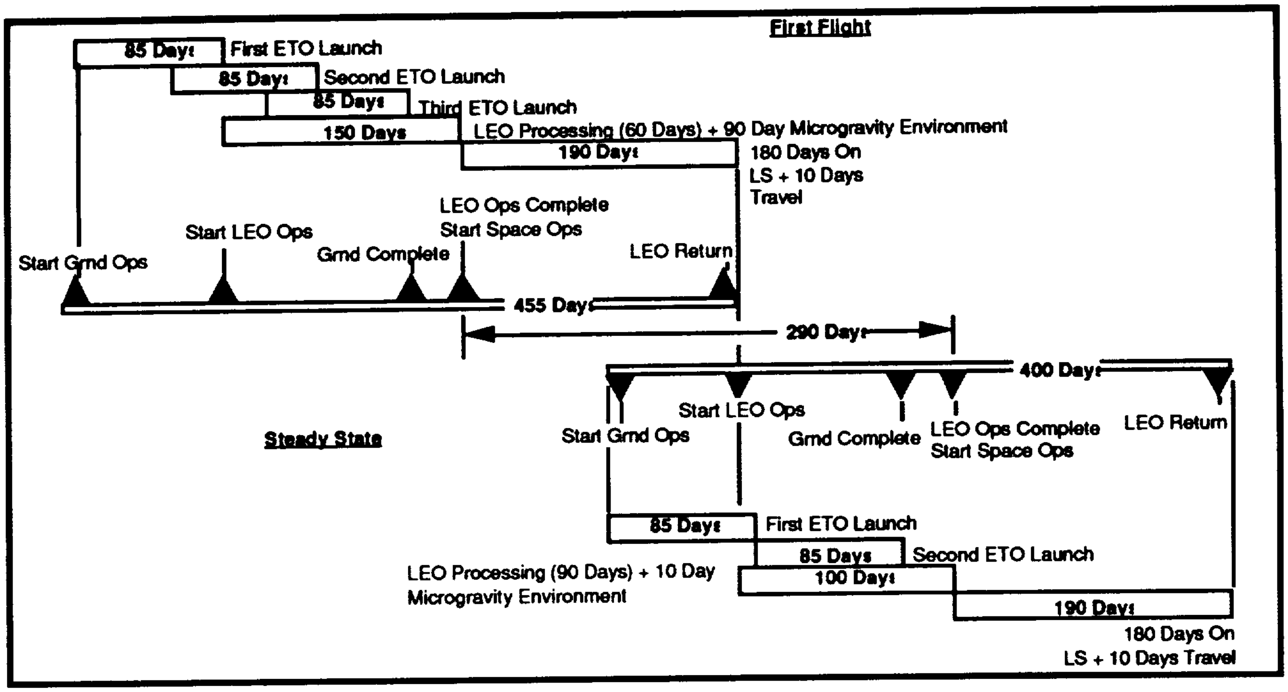


Figure 4.2.2-5: Overall LTS Mission Timeline

Abort Scenarios (outbound mission): The failure modes identified below relate to those mission functions that occur during the outbound phase of a lunar mission. Any abort prior to TLI ignition will result in the vehicle's remaining in low Earth orbit where rendezvous with the orbiting platform can be performed. If rendezvous is not possible, a rescue mission will be conducted to retrieve the crew. A failure of the main propulsion system or GN&C system during the TLI burn which results in an orbit will require a rescue mission to rendezvous with and retrieve the cargo and/or the crew from the disabled vehicle. At some point during the burn the vehicle will pass the point where the RCS system can return it to a safe orbit. In the event of other system failures initiating an abort, the vehicle may continue the TLI burn and establish a free return, slingshot trajectory and return to Earth. If an abort occurs early in the TLI burn the vehicle can return to LEO by using aerobrake or propulsive reentry and await rescue. An abort condition during the separation of drop tanks or first stage will result in a slingshot or free return trajectory beyond the moon automatically returning to Earth where a propulsive reentry to low earth orbit may be performed if the main propulsion system is still operational. If the main propulsion system fails and separation is not completed, the vehicle survivability would be at risk during the aerobrake reentry due to aeroheating of the exposed tanks or stage or increased mass producing excessive heating to the aerobrake. A possible contingency for this would be an EVA for manual removal of the tanks or stage. In the case of a failure of the mid course burn, the vehicle will continue to slingshot beyond the moon and return to Earth where an aerobrake reentry could be performed. If the LLO insertion burn is not accomplished the vehicle will follow the free return trajectory where normal aerobrake reentry could be achieved. An abort initiated during the LLO insertion burn could place the vehicle in an elliptical orbit around the moon. The RCS will be sized to provide sufficient propellant to achieve a lunar orbit from which rescue is possible. A failure during LLO operations requires the vehicle to perform a trans-Earth injection burn and return home, assuming that it has healthy propulsion and navigation systems. Another option would be to continue the mission and achieve a lunar landing where life support equipment will be available as soon as feasible. If the vehicle cannot complete a TLI burn or lunar landing, a LLO rescue mission would be required to rescue the crew. In case of an engine out during landing, the four remaining engines can complete the landing burn. A contingency for this scenario may be to abort to lunar orbit where rescue can be accomplished. A second option is to continue the landing burn and land short of the planned landing site, creating additional problems but not an impossible rescue mission.

Abort Scenarios (return mission): Any abort which precludes launch from the lunar surface will return the crew to the lunar habitation module until a rescue mission arrives. The vehicle has five engines and only two are required for ascent to LLO. Total failure of the navigation or propulsion system during ascent could result in the vehicle's impacting the surface, although if sufficient propulsion can be maintained, the vehicle would be able to accomplish a landing back on the lunar surface at some distance from the lunar base. An abort to lunar orbit would be preferred. An abort initiated during LLO operations requires a trans-Earth insertion burn or a lunar landing burn depending on the nature of the abort to obtain a safe haven for the crew. This depends on the availability of propellants and vehicle status. If the failed system allows the vehicle to return to LEO, the vehicle would return to Earth; if the vehicle remains in LLO a rescue mission is required. A failure of the main propulsion system or GN&C system during the TEI burn resulting in an orbit, will require a rescue mission to rendezvous with and retrieve the cargo and/or the crew from the disabled vehicle. At some point during the burn the vehicle will pass the capability of the RCS system to return it to a safe orbit. In the event of other system failures initiating an abort, the vehicle may continue the TEI burn and establish a free return, slingshot trajectory and return to earth. If an abort occurs early in the TEI burn the vehicle can return to LEO by using aerobrake or propulsive reentry and await rescue. The mid course correction maneuver is normally performed by the main propulsion system. In the event of a failure of this system the RCS is used to accomplish mid course correction. If this approach is followed, sufficient propellant must be retained in the RCS to accomplish aerobrake reentry at LEO. An abort cannot be initiated during aerobrake reentry, once committed, the vehicle cannot turn back. An option might be a propulsive reentry deceleration burn if the vehicle has any remaining main propulsion propellant. This must be sufficient for the entire reentry as a rotational maneuver probably would not be successful in the heat of reentry. An abort initiated in LEO after reentry requires a mission in Earth orbit to rescue the crew.

Abort Scenarios which may occur at any point in the mission: During any failure of the main propulsion system the vehicle uses the RCS to place itself in a rescue attitude; however, this is not possible during all phases of the mission and further study must be done to identify those critical mission phases. If the vehicle has lost propellant to the point that it cannot complete a certain phase of its mission, the cargo or other elements may be jettisoned in order ensure accomplishment of an abort that allows rescue.

The failure of a separation system can be catastrophic to the vehicle depending on the individual system. Failure to separate the TLI drop tanks could result in serious vehicle damage during the reentry. The tanks are not protected by the aerobrake and would burn off damaging the core vehicle in the process. Also the additional mass of these tanks will increase the heating on the aerobrake to the point that it may fail. If sufficient propellant remained in the main propulsion system, a propulsive reentry maneuver would avoid loss of the vehicle and crew. Failure to separate from the aerobrake in LLO would result in an abort whereby the vehicle returns to earth in a normal mode but the mission would be lost. Cargo and propellants may have to be jettisoned in order to satisfy reentry weight limits. Failure of the RCS could result in loss of the vehicle during any of the translational maneuvers such as engine burns, rendezvous, ascent and reentry. Failure of the primary RCS system would initiate an abort of the mission. The electrical power generation, storage and control system is vital during the entire mission; failure of the primary system would result in an abort. The Auxiliary power units (APUs) are used only during engine burns for steering power, failure of one APU initiates an abort. Landing gear consists of legs, pads, shock absorbers and other miscellaneous items supporting the vehicle upon lunar surface landing, failure of this system results in an abort. The nature of the abort and corrective action depends on the point in time the failure occurs. The most probable time for this failure to be detected is in lunar orbit when the landing legs are extended. Abort at that time consists of a trans-Earth burn, returning the vehicle to LEO without landing on the Moon. The caution and warning system automatically checks flight hardware status and alerts the crew to any malfunction. Although it is possible to continue the mission without this system and some of the systems could be monitored by ground personnel, the crew workload would be greatly increased by monitoring the systems manually. A real time decision is required to decide to abort or continue the mission depending on the mission safety guidelines and the point in the mission the system fails. A failure of the caution and warning system is not critical to the completion of the mission; however, if a critical system fails and is not detected, the mission and crew may be lost.

The communications systems are critical throughout the mission. Redundancy allows one individual system to fail without an abort resulting. If several systems are lost and there is no back up the mission is aborted; the nature of the abort depends on the point in the mission that failures occur. Failure of the data processing system results in the abort of a mission. Information from a payload may be required during transit to the moon and the payload might be useless if data is lost. Loss of the GN&C system at any point during the flight could be catastrophic. This system will be adequately redundant however, if the level of redundancy is reduced due to partial failures the mission could be aborted. The abort will depend on the point in the mission at which the failures occur and the individual mission safety guidelines. This system is similar to the caution and warning system in that it monitors the health of the flight vehicle among other items. If the health of the flight vehicle cannot be adequately determined the mission safety guidelines call for an abort. This abort will depend on the particular data missing, and the impact of the loss of that data assuming worse case (the system or component being monitored fails and failure is not detected). The nature of the abort will depend on the point in the mission the abort is initiated.

It is impossible to determine all the failures or combination thereof that could occur during a mission of this complexity and duration; however, through redundancy of systems and a detailed analysis of failures, it is possible to show that the crew and payload should be able to reach safe haven at any point during the mission.

4.3 Surface Operations

The LTS operations on the lunar surface are limited to cargo and crew loading and unloading, station-keeping monitoring, and unscheduled maintenance of mission critical elements. This section discusses the loading and unloading of crew and cargo; monitoring and maintenance have been discussed as part of the space flight operations, Section 4.2.2. The current scenario for the delivery of cargo and crew to the lunar surface is based on the requirements of the "Option 5" SEI lunar outpost. Figure 4.3-1 defines the manifesting of the cargo over the life of the outpost program. This manifest identifies "cargo only" missions, of which three of the four required are to be configured in a cargo reusable mode instead of expending the system at the surface. Piloted missions deliver a crew of four along with cargo. Figure 4.3-2 depicts a typical cargo configuration at lunar landing. Shown is how payloads can be manifested atop the cargo platform to maintain cg control. The cargo identified in this figure represent the second lunar flight (designated Flight 1) which consists of the lunar habitat module, a power module, an airlock, and a half cargo pallet. Total manifested mass for this flight is 26.3 tonnes. Although this mass is above the current performance for a reusable cargo mission, some remanifesting of other lunar mission

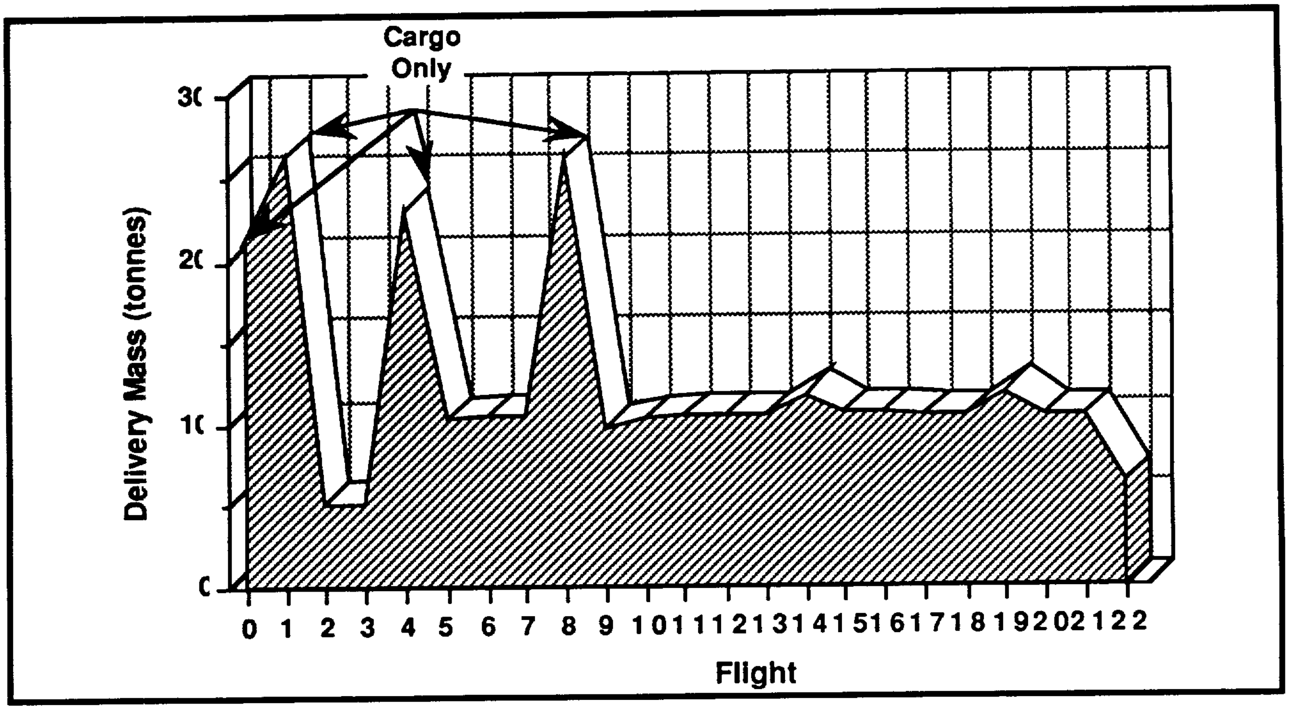


Figure 4.3-1: Lunar Outpost Cargo Manifest

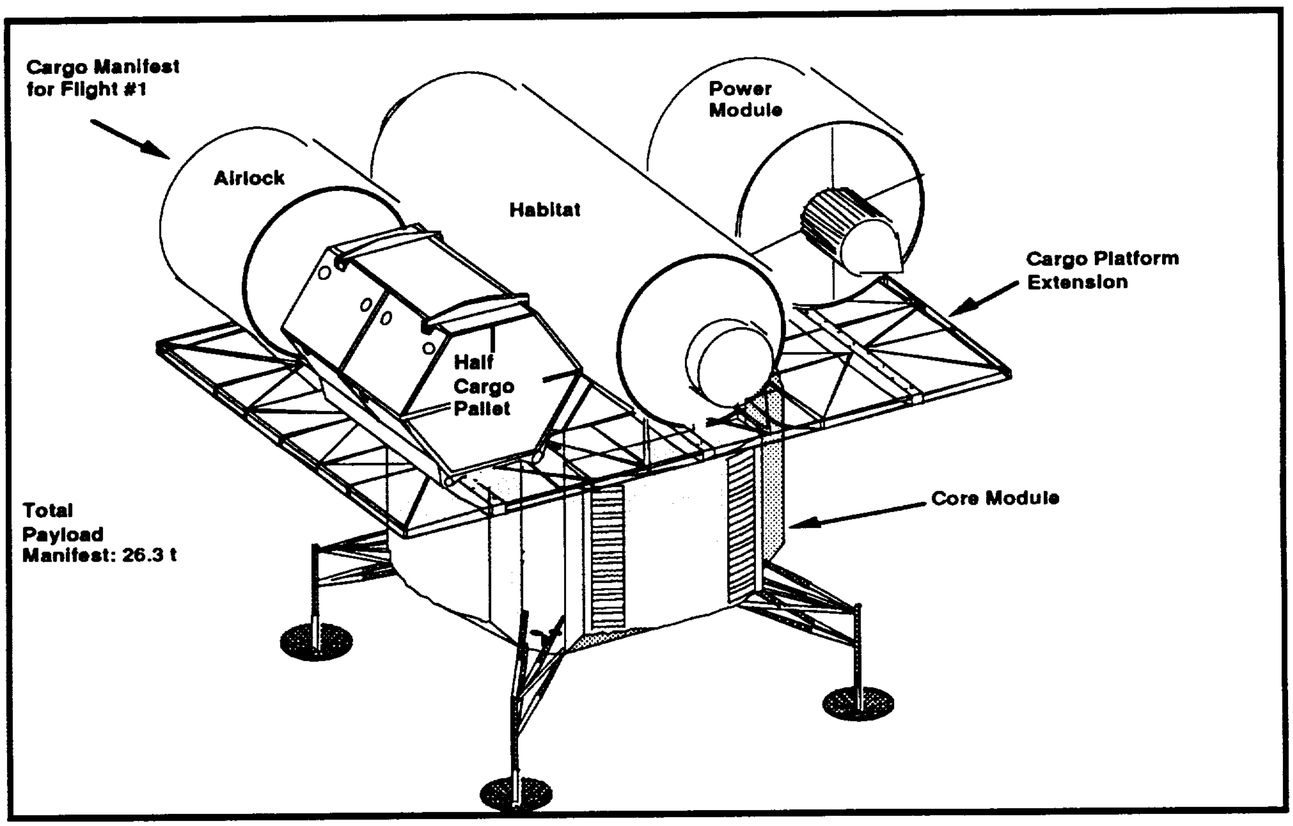


Figure 4.3-2: Typical Cargo Configuration at Lunar Landing

would bring this cargo in line with vehicle capabilities. Figure 4.3-3 depicts the piloted configuration at lunar landing. The drop tanksets have been released after the TLI and LLO burns. The aerobrake and its associated equipment have been left in LLO. The landing vehicle consists of the core module with the crew module and the lunar cargo. Total manifested mass for this flight is < 14.6 mt.

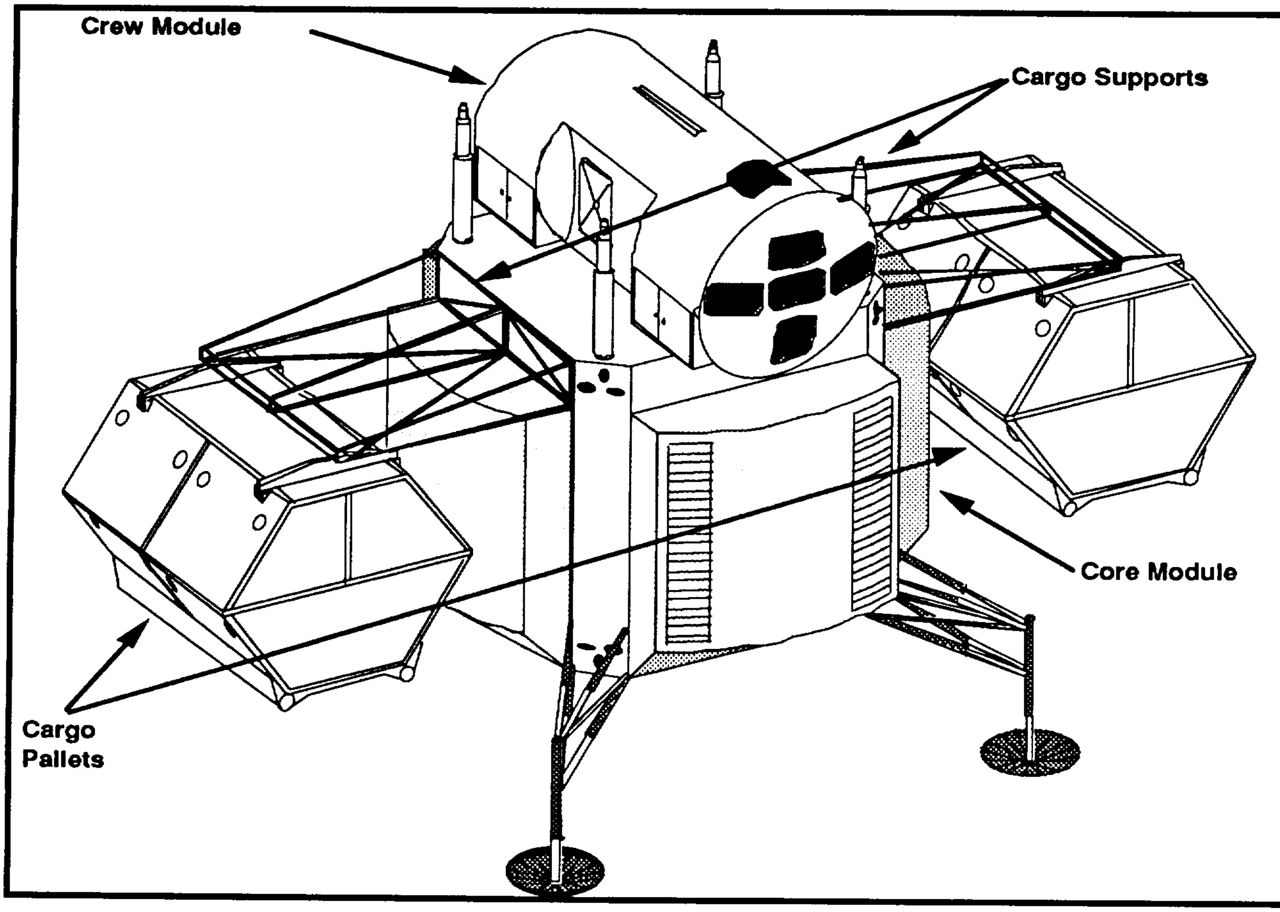


Figure 4.3-3: Typical Piloted Configuration at Lunar Landing

Once the cargo has been delivered, it must be unloaded by surface support equipment or by the LTS to transportation equipment. Because deliveries are made in both cargo and piloted configurations, both unloading systems will be used. The large cargo platforms require the availability of surface loading/unloading equipment, as unloading these platforms is not feasible with the current piloted system configuration. This surface unloader/loader has been defined as the Lunar Excursion Vehicle Payload Unloader (LEVPU) by Planetary Support Systems (PSS) inputs to the "Option 5" SEI Lunar Outpost Initiative. Figure 4.3-4 shows the LEVPU unloading cargo from the cargo configuration on the lunar surface. The vehicle configuration is sized to allow the payload unloader to roll over and straddle the vehicle and its cargo. Once positioned over the vehicle, the unloader picks up a piece of cargo, lifts it, and proceeds to roll away from the vehicle. After the cargo has been deposited in its position on the lunar surface or on a transporter, the unloader proceeds back to the vehicle to unload another piece of cargo. Cargo unloading of the piloted vehicle on the Lunar surface can be accomplished without the use of the LEVPU, as shown in Figure 4.3-5. The cargo is supported by supports extending from the sides of the core. Once the vehicle has landed on the lunar surface, the cargo can be lowered directly to the surface or onto a transporter by using a hoist mounted on the cargo support structure. These hoists (Fig. 4.3-6) allow cargo to be lowered directly to the lunar surface. The spacing between the legs of the core allows the cargo to be lowered directly to the surface.

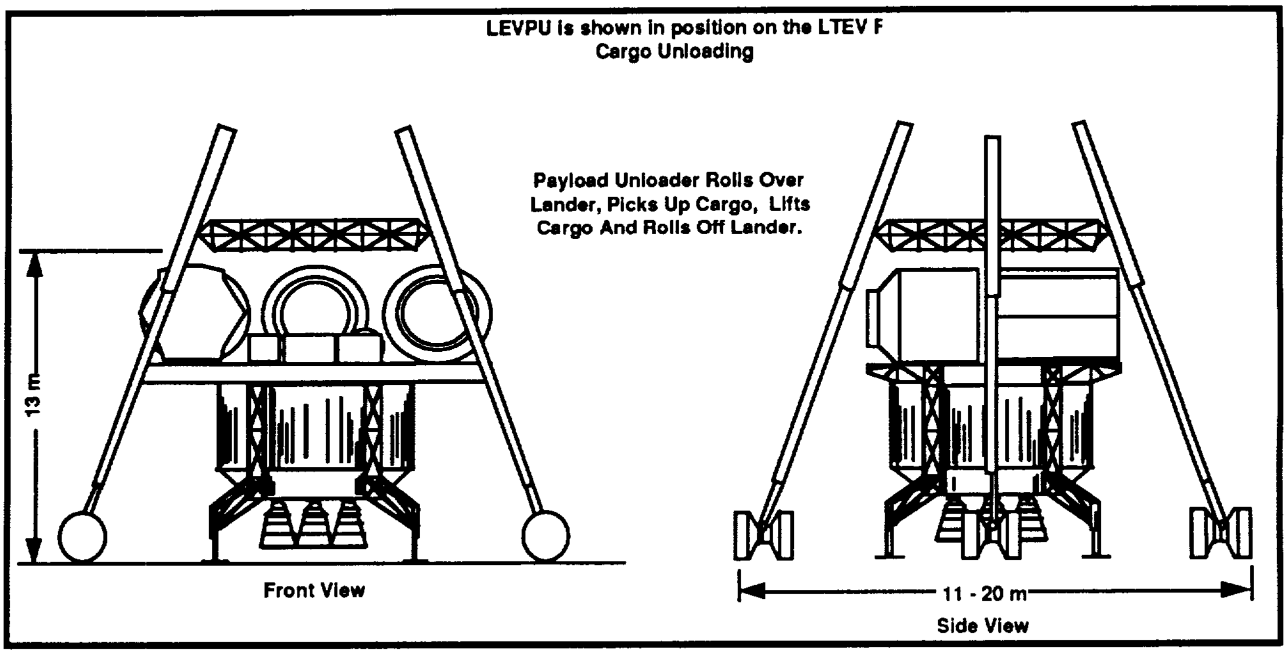


Figure 4.3-4: LEVPU Unloading Cargo on Lunar Surface

After landing, connection of the surface umbilicals for transferring of propellant and data management will be made by surface support equipment. Details of this function as well as the equipment to conduct it, have not been defined at this time; however, it is known that the interfaces to the LTS will be compatible with those used at SSF and KSC. Details of specific interface requirements will be discussed in section 4.4.

4.4 Interfaces

The LTS will interface with several of the primary space infrastructure elements during the execution of a single lunar mission. These elements include the ground processing facilities at KSC, the ETO system during transport into LEO, SSF during assembly, verification, and

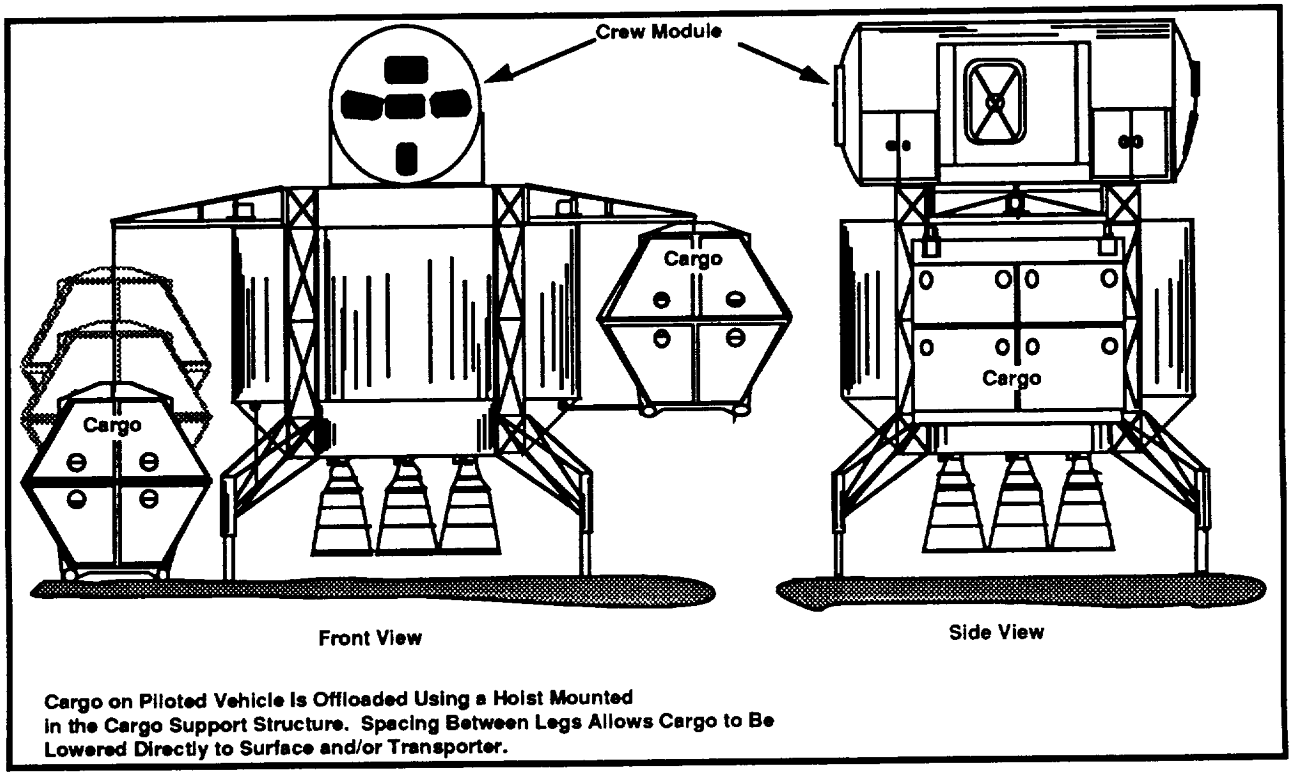


Figure 4.3-5: Piloted Vehicle Unloading Cargo on Lunar Surface

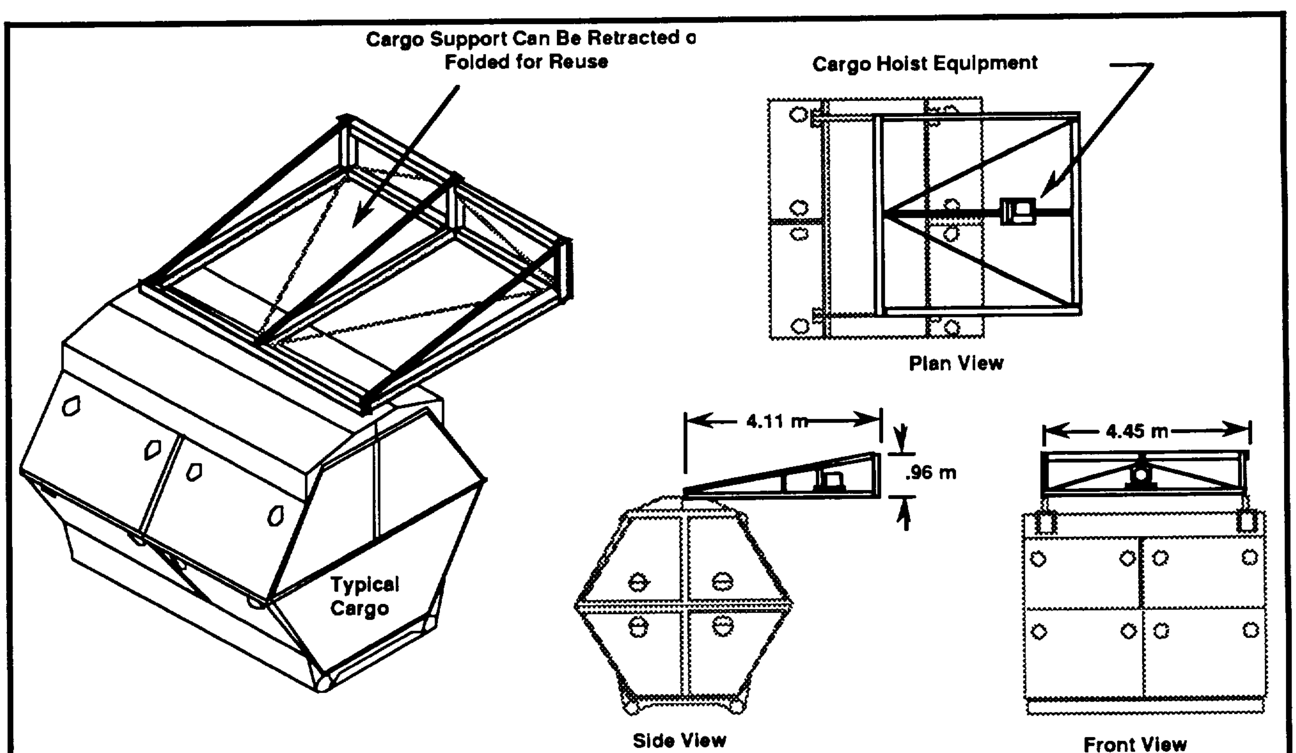


Figure 4.3-6: Cargo Unloading Hoist

refurbishment, PSS cargo during transfer between LEO and the lunar surface, and the lunar outpost facilities throughout the duration of the surface stay time. Discussed in this section will be the principle interfaces as defined for each of these support nodes.

The STV interfaces for both ground processing and the HLLV are identified in Tables 4.4-1 and 4.4-2. Envelope dimensions indicate the handling size but do not include accessibility requirements or GSE allowances. Vertical transporters, handling dollies, and tractors are required for each of the STV modules and require (or share) an HLLV payload shroud vertical transporter. Electrical power will interface with the ground system only during stand alone processing in the SPIF using drag on cables. During other phases of processing power that is required will be provided through the ASE from the HLLV. All fluids (liquids & gasses) are loaded during stand alone processing in the SPIF except for cryogenic propellants (LOX & LH2) which are the only fluids loaded at the launch pad. The HLLV shroud containing the STV elements will be purged at the launch pad. Active EC System support by GSE is required only in the SPIF for crew module stand alone processing. There are no unusual safety considerations common with all cryogenic space/launch vehicles such as handling of propellants/cryogenic fluids at the launch pad, high pressure gasses, oxygen deficiency in the crew module at the SPIF, and handling/transfer of loads overhead that have been considered. Security will be consistent with NASA/DOD programs. There will be no physical pin/plug interfaces between the STV modules and the ground system in areas of ground processing except for the instrumentation through the HLLV payload shroud ASE. All communication, test control, command control, instrumentation data, etc., are through links not requiring physical interfacing, such as RF, IR, and optical links. A low positive pressure is maintained on tank modules assuring the maintenance of structural and cleanliness integrity.

Because SSF conducts many of the same types of functions performed at KSC, similar interfaces are found. These interfaces provide an unpressurized area which provides meteoroid protection and active and passive thermal control for the STV. A teleoperator manipulator dedicated to STV is planned along with an interface with SSF electrical power. Communications and tracking are provided by SSF for the monitoring of critical operations and support of overall mission functions. The STV Environmental Control and Life Support Systems (ECLSS) interface with and are supported by an SSF module. All cryogenic fluids will be supplied from Earth and not from SSF. Proximity operations will be controlled from SSF. Personnel transfer to and from the STV assembly area are provided by SSF. Tables 4.4-3, 4.4-4, and 4.4-5 show these interfaces along with the functional relationships to the assembly, check-out and refurbishment phases of the LEO activities.

Table 4.4-1: KSC Ground Processing Interfaces

Interface	Core	Crew Module	Aerobrake
Envelope	8.5 m	3.7 m dia x 8.5 r	8.5 m dia envelop
Handling	hooks & fittings to I/F w/ vertical transporter dollies and tractors	hooks & fittings to I/F w/ vertical transporter dollies and tractors	hooks & fittings to I/F
Electrical	drag on cables - SPIF ASE thru HLLV on pad	drag on cables - SPIF ASE thru HLLV on pad	drag on cables - SPIF ASE thru HLLV on pad
Mechanical	Handling-Gnd / HLLV	Handling-Gnd / HLLV	Handling-Gnd / HLLV
Propellants	N/A	Life support fluids loaded in SPIF	NA
Pneumatics	loaded in SPIF	loaded in SPIF	loaded in SPIF
Environmental Control	HLLV shroud purg	HLLV shroud purg	HLLV shroud purg
Safety	High pressure gasses cryo handling	High pressure gasses cryo handling	No unusual safety requirements
Security	normal NASA requirements	normal NASA requirements	normal NASA requirements
Communications	ground I/Fs thru fiber optical, RF or IR links	ground I/Fs thru fiber optical, RF or IR links	ground I/Fs thru fiber optical, RF or IR links
Cabling	electrical and instrumentatio	electrical and instrumentatio	electrical and instrumentatio
Operational Constraints	Pos pressure on tank Maintain clean syster	Pos pressure on tank Maintain clean systen Maintain cabin ai	Pos pressure on tank Maintain clean systen

Table 4.4-2: KSC Ground Processing Interfaces

Interface	TLI/LOI Tanks	Return Pallets
Envelope	4.6 m dia x 8.7 m ea.	4.6 m x 2.7 m x 2.6 m (pallet)
Handling	hooks & fittings to I/F w/ vertical transporter dollies and tractors	hooks & fittings to I/F
Electrical	drag on cables - SPIF ASE thru HLLV on pad	drag on cables - SPIF ASE thru HLLV on pad
Mechanical	Handling-Gnd / HLLV	Handling-Gnd / HLLV
Propellants	filled thru umbilicals on HLLV shroud	filled thru umbilicals on HLLV shroud on pad
Pneumatics	loaded in SPIF	loaded in SPIF
Environmental Control	HLLV shroud purge	HLLV shroud purge
Safety	High pressure gasses cryo handling	High pressure gasses cryo handling
Security	normal NASA requirements	normal NASA requirements
Communications	ground I/Fs thru fiber optical, RF or IR links	ground I/Fs thru fiber optical, RF or IR links
Cabling	electrical and instrumentation	electrical and instrumentation
Operational Constraints	Pos pressure on tanks Maintain clean system	Pos pressure on tanks Maintain clean system

Table 4.4-3: SSF/LTS Interfaces

Interface	Structural	Mechanical	Environmental	Electrical Power	Data Mgmt
Interface Description	Enclosure based support structure (details TBD)	Teleoperated manipulator end effectors	Enclosure debris shielding (TBD protection); Contaminant collection & venting	DC to DC Converter Unit (DDCU) (6.25 kW, 120 VDC); Secondary Power Dicstribution (1-130A or 2-50A or 4-25A or 8-10A)	FDDI Protocol (100 MBPS CCSDS packet format); 20MHz NTSC Video Downlink
Mission Phase					
Hardware Delivery	•	•	•		
Assembly	•	•		•	•
Verification	•			•	•
Propellant Servicing	•	•	•	•	•
Closeout	•			•	•
Deployment/Launch	•	•	•		•
Retrieval	•	•	•		
Refurbishment	•			•	•

Table 4.4-4: SSF/LTS Interfaces

Interface	Thermal Control	Comm & Track	EVA	ECLSS	Man Systems
Interface Description	Cold plates (62F & 35F); 6.25 kW heat rejection per heat exchanger	UHF Space-to-Space radio	One 2-Man EVA event per day (6 hours per event)	14.7 psi, 79% N2, 21% O2 atmosphere at 50% humidity	TBD
Mission Phase					
Hardware Delivery		•			
Assembly	•	•	•		
Verification	•	•	•	•	•
Propellant Servicing	•				
Closeout	•		•	•	•
Deployment/Launch		•		•	•
Retrieval		•			
Refurbishment	•		•	•	•

Table 4.4-5: SSF/LTS Interfaces

Interface	Fluid Mgmt	Proximity Ops	Transfer Ops
Interface Description	175 psia N2 supply; Water supply; Waste gas disposal.	TBD	TBD
Mission Phase			
Hardware Delivery		•	•
Assembly	•		•
Verification	•		
Propellant Servicing	•		•
Closeout	•		
Deployment/Launch		•	•
Retrieval		•	•
Refurbishment	•		

During transportation of the crew and cargo, or just cargo to and from the lunar surface, interfaces between the LTS and the cargo exist. To minimize the impact to the LTS, the interfaces shown in Table 4.4-6 include only the physical attachments of the cargo to the vehicle and electrical to provide monitoring of the health cargo itself. Handling attachments for placing the cargo on the STV will be provided by the cargo. No liquid or pneumatic interfaces will be supplied by the STV to the cargo although minimal electrical power for monitoring and statusing is provided. Environmental control and meteoroid protection, if required, is supplied by the cargo. Communications support will be provided by STV for health and status monitoring only.

After the LTS arrives on the lunar surface, key interfaces (Table 4.4-7) are required to ensure the vehicle's return to LEO at the completion of the lunar stay and the unloading of cargo in support of lunar outpost. The LEV servicer is moved and positioned on the lunar surface by the LEVPU, providing continuous electrical power to the STV lander for 48 hours after descending to the lunar surface. Mechanical interfaces include a support for the environmental shield. The LEV servicer provides cryogenic reliquefaction of propellants pending development environmental shield. The LEV servicer provides cryogenic reliquefaction of propellants pending development of a system for the generation and replenishment of cryogenics on the lunar surface. Environmental control support provided by the LEV servicer includes thermal control and protection from lunar ejecta. The LEV servicer provides the necessary communications to monitor critical systems and support all mission requirements.

Table 4.4-6: Cargo/LTS Interfaces

Interface	Cargo
Envelope	N/A
Handling	mechanical attach to core vehicle
Electrical	interface with STV for electrical power
Communications & Tracking	health and status monitoring only
Environmental Control	provided by cargo
ECLSS	N/A
Liquids / Pneumatics	N/A
Prox Ops	N/A
Personnel Transfer	N/A

Table 4.4-7: PSS/LTS Interfaces

Interface	PSS Servicer
Envelope	N/A
Handling	moved and positioned by LEVPU
Electrical	provides continuous electrical power to lander after 48 hours
Communications & Tracking	monitor critical systems support mission rqmts
Environmental Control	provides thermal and lunar ejecta protection
ECLSS	N/A
Liquids / Pneumatics	provides lander cryogenics reliquifaction possible cryogenic replenishment
Prox Ops	N/A
Personnel Transfer	N/A

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5.0 PROGRAMMATICS

5.1 PROJECT PLANNING AND CONTROL INTRODUCTION

During the initial phase of the Space Transfer Vehicle Concepts and Requirements Study contract, the project planning, finance, and data management activities were combined into a single functional task. This task provided management with the tools required to control the business management aspects of the contract. The study plan (DR-1) was updated after negotiations, submitted and approved by NASA/MSFC. This study plan was then used to monitor program schedule and cost performance. The STV Study Program Master Schedule (Fig. 5.1-1) and program technical status were then reported to NASA/MSFC in the monthly program progress report (DR-3). The monthly program financial status was reported to NASA/MSFC via the NASA form 533M, and an estimate to complete was provided to NASA/MSFC on a quarterly basis in the NASA form 533Q.

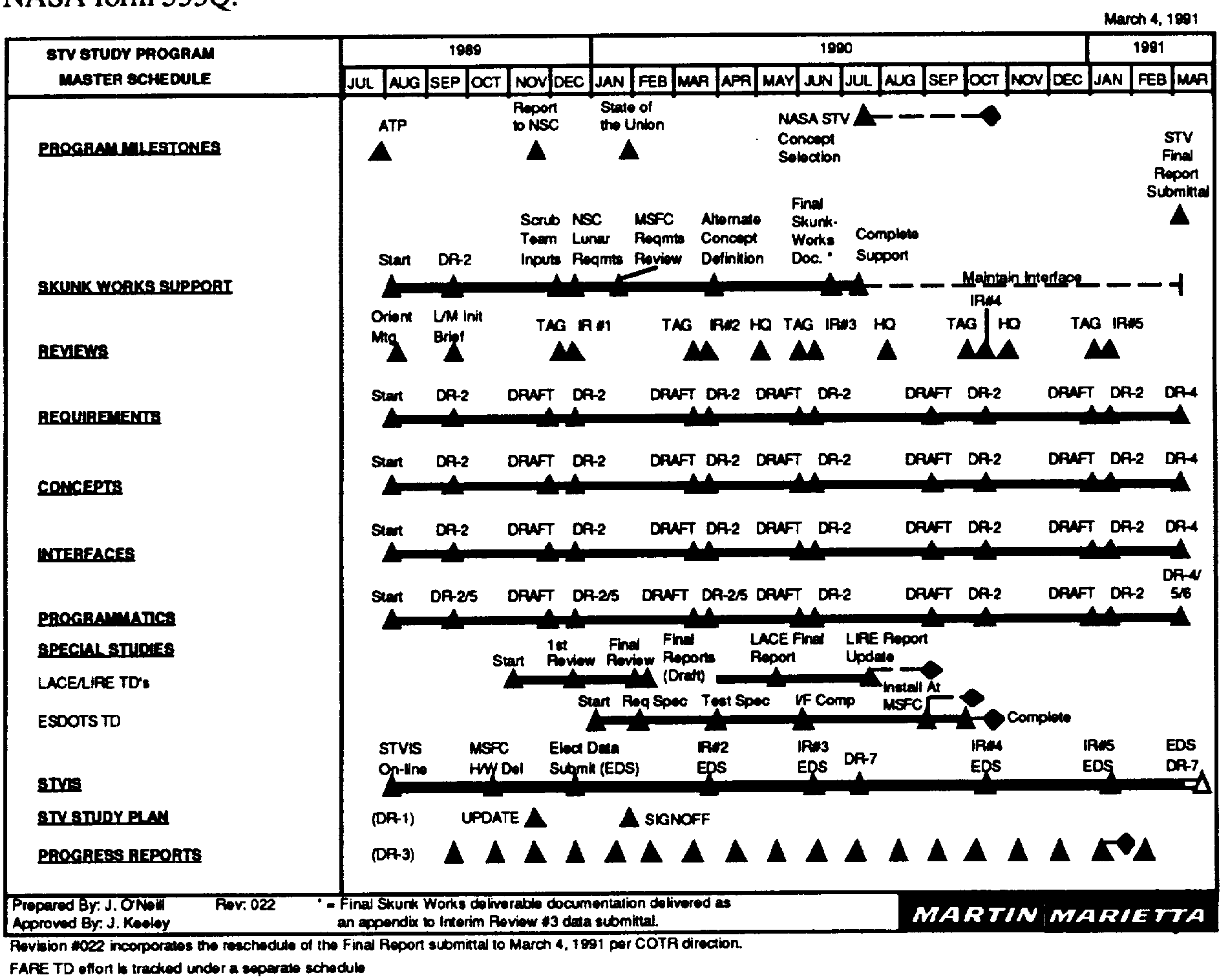

Figure 5.1-1 STV Study Program Master Schedule

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The development of the summary phase C/D and phase E/F planning data was accomplished during this study phase. Based on the direction taken in the Space Transfer Vehicle (STV) basic defined tasks contract activities, detailed project logic network models were developed for the Lunar Transportation System (LTS) as the major emphasis and STV programs. The network models have been developed to the subsystem level based on the current depth of conceptual maturity, and are directly traceable to the major work breakdown structure (WBS) element. Both the required critical path analysis and risk assessments have been accomplished and are documented in this final report. Incremental delivery of the project planning data has been accomplished with inclusion in the performance review documentation (DR-2) submittals at the quarterly Interim Review (IR) meetings held at NASA/MSFC.

5.1.1 Summary of Approach

The Martin Marietta approach in designing the program plan for the STV DDT&E program has been to implement a technique of database development based upon a foundation of experience, history, and lessons learned from contracts of comparable size and/or complexity. Our integrated scheduling system uses a hierarchical structure combining the use of computer data bases and electronic interfaces tying the program master schedule with detailed lower tier schedules such as engineering release plans, master build plans, subcontract/material delivery plans, and test plans. Using a flow down approach, the contract master schedule has been established to reflect contractual requirements, major program interfaces, and significant program activities. The next tier of the hierarchy was the intermediate schedules which bridge the contract master schedule to the lower level detail schedules. The STV/LTS network logic models have been developed at this intermediate level and are used to validate the contract master schedule. This database would also be the vehicle for flow down of all schedule requirements to the lower level detail schedules upon the initiation of a DDT&E program.

Our STV network logic models were created for each of the major WBS levels (refer to the DR-5 submittal of the WBS and its associated WBS dictionary) and then input to the Open Plan™ network database (PC based). Inter-relationships were established and activity durations derived from historical data and were then adjusted from experience and lessons learned arriving at the expected measure of time to perform each activity. All design and fabrication activities are based on the Martin Marietta Denver calendar of five-workdays per week at eight hours per day. The KSC processing, LEO processing, and mission ops activities are based upon a seven-day week at varying hours per day dependent upon the tasks to be performed and their respective locations.

Through this process we were able to analyze the program model to determine critical paths and identify areas of potential risk.

5.1.2 Groundrules and Assumptions

The program plan has been established using MSFC and Martin Marietta mutually agreed upon groundrules and assumptions. The initial coordination of these groundrules and assumptions took place at the STV programmatics discussion meeting held at NASA/MSFC on 14 February 1990. The basic groundrules have remained the same throughout the project plan development, with the exception of the Initial Launch Capability (ILC) for the STV and LTS. The driving factor in the movement of the ILC dates was its direct relationship to the program phase C/D authority to proceed (ATP). The following are a list of groundrules that were used in the development of the STV program plan:

- Addresses Program Phases C/D/E/F
- Developed For The Selected STV Concept Configuration
- Developed To The Subsystem Level and Dependent Upon Conceptual Maturity
- Traceable To The WBS (Major WBS Element and Subsystem)
- STV ILC To Be Achieved In 2001 (Was 1998)
- LTS ILC To Be Achieved In 2003 (Was 2001)

The Martin Marietta assumptions were developed to further guide the direction of the program plan. Development of the assumptions assisted in establishing boundaries to the plan thereby making the task achievable in this program phase. The following are a list of assumptions that were used in the development of the STV program plan:

- Plans Baselined To Option 5 Schedules (Option E In The 90 Day Report)
  - Reference MSFC Schedules: Space Transfer Vehicle Dated October 1, 1990 (S. Spearman); Assumptions For FY 91 Budget Planning Dated December 12, 1990 (N. Chaffee); Heavy Lift Launch Vehicles December, 1990 (A. Jackman)

- STV Program Plans Developed For:
  - STV As An HLLV Upper Stage (Development In Parallel With HLLV Development Program With An STV Phase B ATP Required By HLLV PDR)
  - Phased Progression To The Lunar Transportation System (LTS)
- ETO Vehicle Flights Assumed Available To Support STV Program Plans

•Unmanned STV Test Flights Incorporate Man-rating Demonstrations
•STV "Real Payload" Deliveries Incorporate Test/Confidence Objectives
•Manned Elements - Minimum Of Three Unmanned Flights Before Crew Flight, (Re Apollo)
•Long Lead Procurement Assumes Low Risk Materials To Mitigate Schedule Risk
•Tooling Assumed Available To Support The Program Schedules
•No Interference In Facilities During Fabrication, Assembly, Test, Processing, and Launch
-New Facilities Will Be On-line To Accommodate The Program Schedules

5.1.3 Summary Master Schedules

The HLLV/STV Program Schedule (Fig. 5.3.1-1) illustrates the interrelationship between the HLLV development program and the development program of an STV/HLLV upper stage. The HLLV schedule data reflects the sequencing of the anticipated major milestones for PDR, CDR, and test flight. The schedule then shows the time phasing requirements to implement an almost parallel program for an STV as an HLLV upper stage with the phased progression to the Lunar Transportation System (LTS). The fifteen foot diameter STV schedule is included to accommodate the interface for the Space Shuttle, an upgraded Titan IV, or other fifteen foot diameter payload class of vehicle as identified in the STV statement of work. The STV schedule for the fifteen foot diameter and the HLLV upper stage meets the early IOC dates for the NASA polar mission and the DoD missions from the CNDB-90. These STV systems are in service while the development of the LTS progresses through the first test flight launch in 2003. An expendable LTS cargo mission (payload unloader) to the lunar surface follows in 2004 and a reusable LTS cargo mission and the first piloted mission in 2005. This program phasing lowers peak funding requirements and provides integration of the mature STV design into the LTS. This sequencing also increases the ability to use common test beds and previous STV test articles through modifications and upgrades for LTS scenarios (schedule permitting) and provides early flight mission confidence using the STV prior to the LTS flights. The early STV flights will accomplish selected LTS test objectives and lower the development time, cost, and risk for the LTS program.

Our STV Summary Master Schedule (Figs. 5.3.1-2 and 3) presents an overview of the program plan for the fifteen foot transfer vehicle and depicts the effort from phase C/D authority-to-proceed in October 1995 to full scale development and on through the beginning of the production phase.

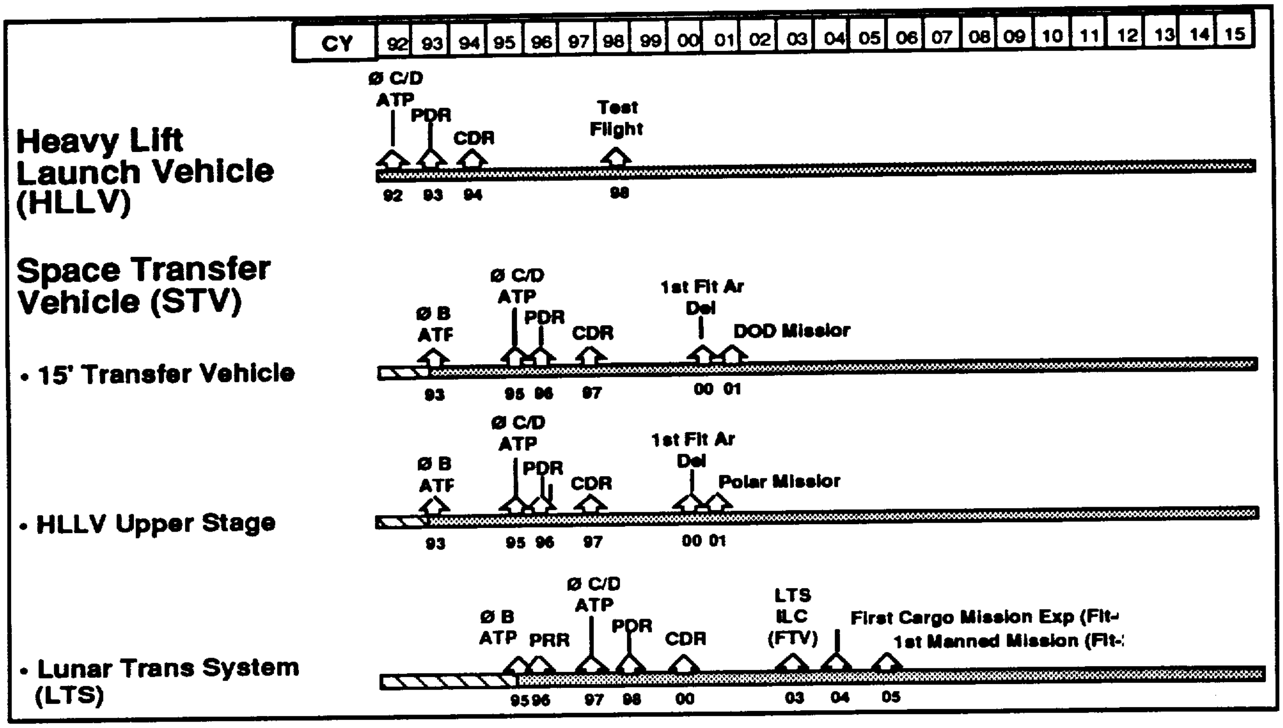

Figure 5.1.3-1 HLLV/STV Program Schedule

Critical milestones identified by NASA/MSFC and Martin Marietta are presented. It should be noted that there is an overlap between production and FSD which extends through ILC. In order to mitigate risk in reaching ILC on schedule, long lead procurement approval for low risk materials which are needed in the development hardware builds will be required at Program Requirements Review (PRR) in the January 1996 timeframe. Also, an additional long lead procurement authorization to support production hardware builds will be required at the completion of Critical Design Review (CDR). The primary and secondary critical paths are also highlighted on the summary schedule. The primary critical path as derived from the logic network model is traced through the design, development, and qualification of the main propulsion system (which includes the advanced space engine and RCS system) leading up to integration into the first flight unit. This portion of the database was generated using the anticipated Integrated Modular Engine (IME) development program and SSME historical data to arrive at the expected task durations for engine development program and historical data from Viking, Magellan etc., for the RCS program development task durations. The secondary critical path is annotated and, in the event that an existing version of an off the shelf (three year lead time) RL10 engine is used on the front end of the program to cut costs and reduce risk, then the primary path would shift to the avionics development program, in particular the GN&C and Data Management/Sequencing subsystems and their associated software development programs.

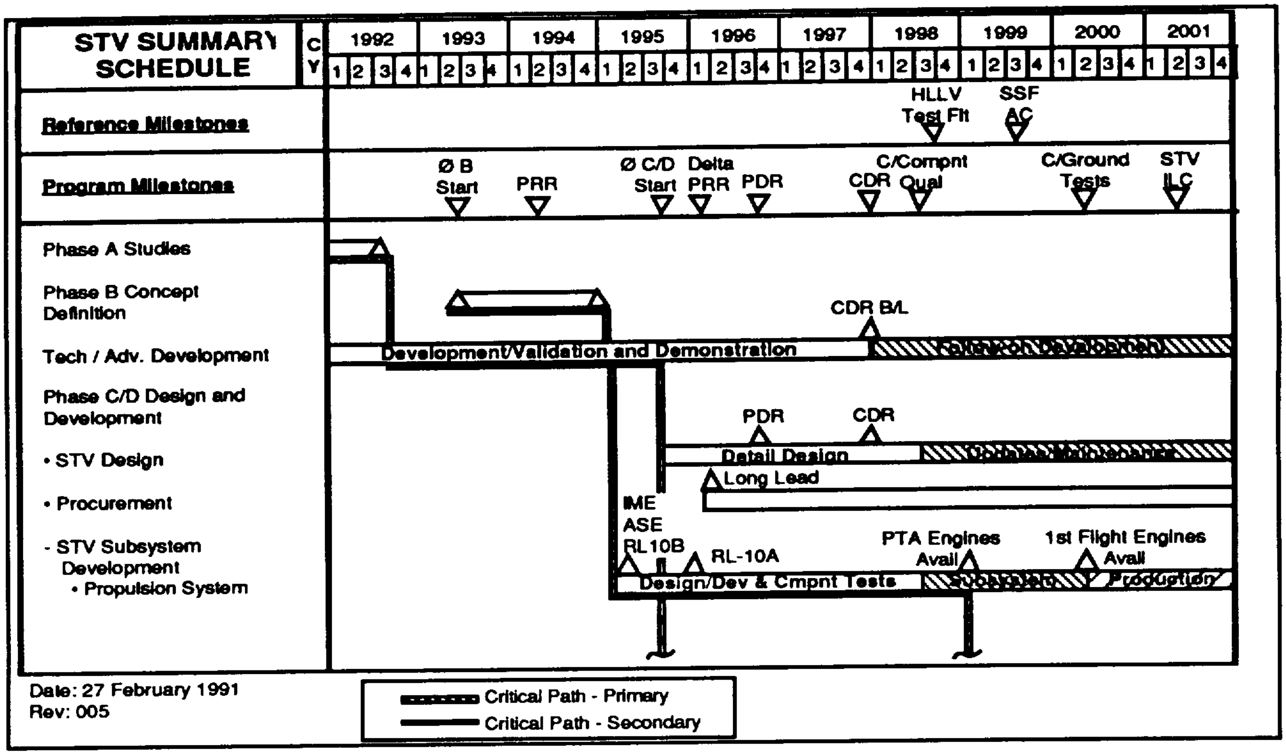

Figure 5.1.3-2 STV Master Summary Schedule

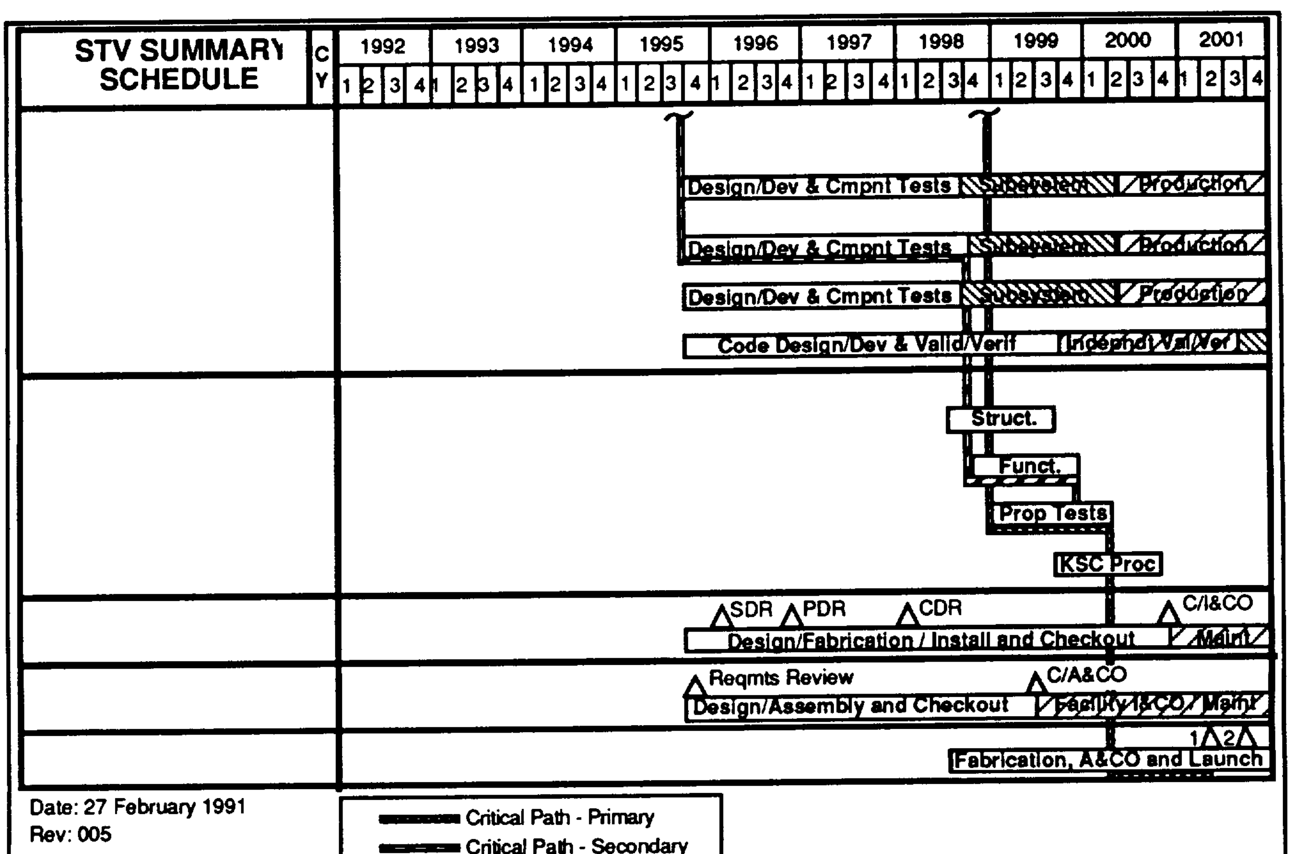

Figure 5.1.3-3 STV Master Summary Schedule (Cont'd)

Our LTS Summary Master Schedule (Figs. 5.3.1-3 and 4) presents an overview of the program plan for the selected space based reusable concept 4E-5B and shows the effort required from the phase C/D authority-to-proceed in October 1997 to full scale development and on through the beginning of the production phase. The database was generated using historical data, lessons learned and experience from the Viking, Magellan, MMU, Titans, Skylab, TOS, USRS, Transtage, FTS, TSS, Apollo, etc., programs. Critical milestones as identified by NASA/MSFC and Martin Marietta are presented. It should again be noted that there is an overlap between production and FSD which extends through ILC and in order to mitigate risk in reaching ILC on

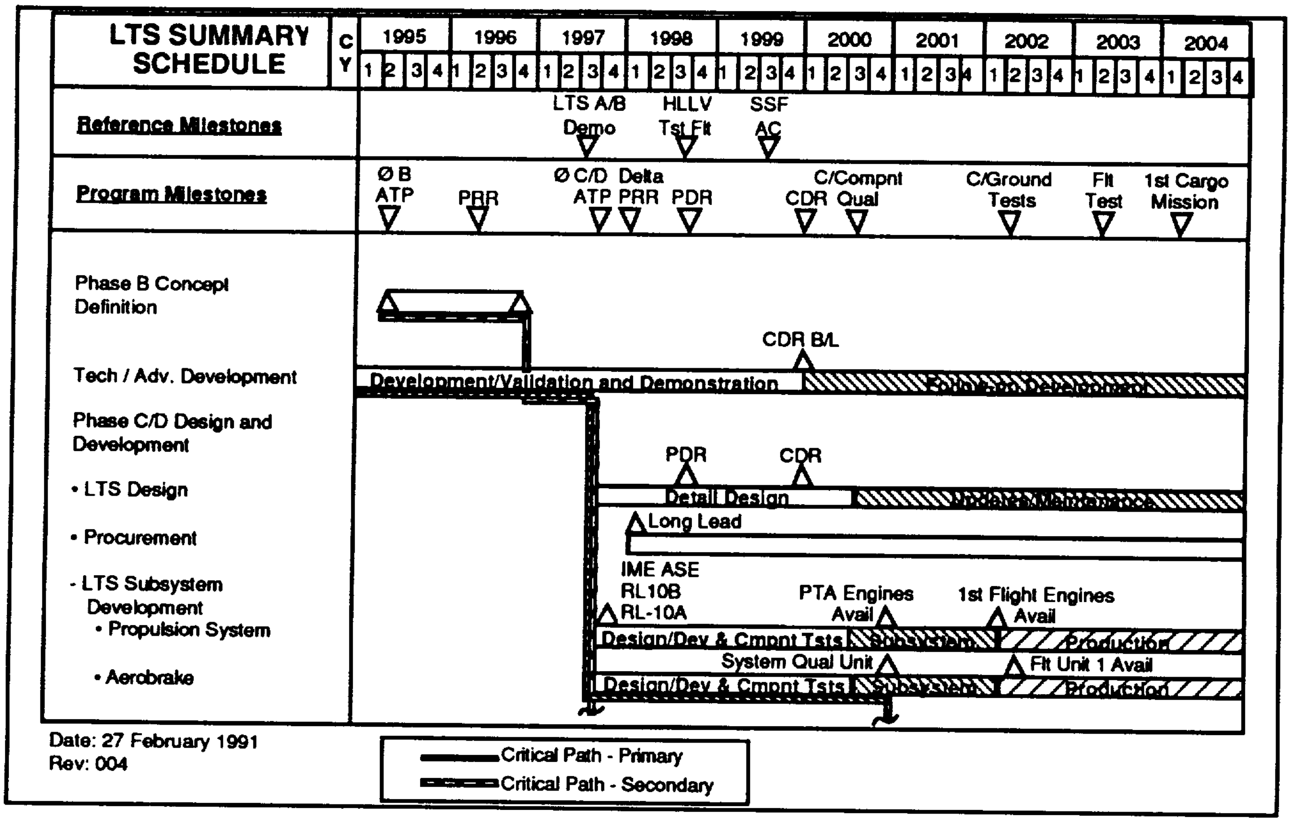

Figure 5.1.3-3 LTS Master Summary Schedule

schedule, long lead procurement approval for low risk materials needed for the development hardware builds will be required at Program Requirements Review (PRR). Also an additional long lead procurement authorization to support production hardware builds will be required at the completion of Critical Design Review (CDR). The primary and secondary critical paths are also highlighted on the summary schedule. The primary critical path as derived from the logic network model is traced through the development and ground/flight qualification of the smart aerobrake leading up to integration into the first flight unit. The smart aerobrake required the development of the integral avionics, RCS, and structural package into a development program that meets the early mission objectives and supports the defined LTS test program. The secondary critical path is annotated through the avionics development program, in particular the GN&C and Data Management/Sequencing subsystems and their associated software. The complexity of the crew module and the length of an ASE development program combined for a small margin of difference between all four programs and either of them could become the program plan tallpole.

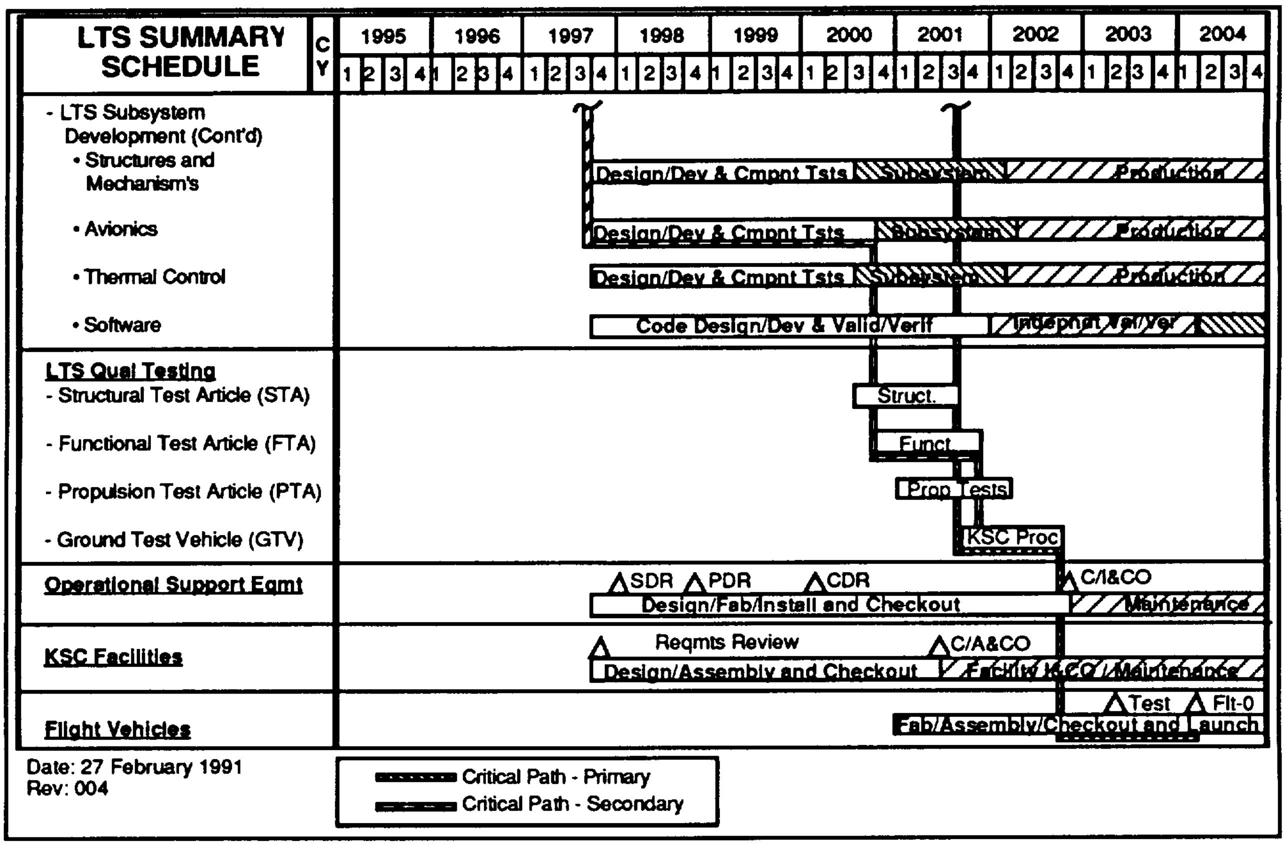

Figure 5.1.3-4 LTS Master Summary Schedule (Cont'd)

5.2 TEST PROGRAM DETAILS

The STV/LTS test program has been developed to show an integrated approach of satisfying both the component and system test requirements of the ground and flight articles. To assure the success of this test program it has been divided into test phases which parallel the STV/LTS program phases B, C/D, and E/F. The following briefly describes each of these phases and the test intentions: a) technology verification and feasibility of STV/LTS design concepts during phase B; b) design development testing during phase C/D; c) component and system qualification program during phase C/D; d) systems level ground and flight testing during phase C/D; and e) acceptance and operational testing during phase C/D and phase E/F.

The test program groundrules and assumptions were developed as top level boundaries based upon the accepted policies and procedures of the Martin Marietta Corporation. The following are a list of assumptions that were used to generate the preliminary test plan for the STV/LTS:

- The system will be qualified to MIL-STD 1540B
- Components will be qualified to M-67-45, Test Methods and Controls, Components
- A full static firing test of the STV/LTS will NOT be required
- Aerobrake undock/rendezvous/docking will be demonstrated in LEO
- An unmanned, full-scale aeropass maneuver will be tested in space prior to the first piloted aeropass maneuver
- An unmanned crew module will be tested in space prior to the first piloted mission to the lunar surface
- Test articles are assumed to have a single use only
- Near earth missions can/will be utilized to accomplish lunar test objectives

The following are a list of assumptions that were used to develop the STV/LTS test program schedule and network logic model:

- The LTS test flight will be accomplished in 2003
- The first STV as an HLLV upper stage mission (polar mission) will occur in 2001
- The first STV as an 15' transfer vehicle mission (DOD) will occur in 2001
- The first LTS cargo mission (Flt-0, payload unloader) will occur in 2004
- The first LTS piloted mission (Flt-2) will occur in 2005

The STV/LTS phase B ground testing scenario has been established to provide technology verification and feasibility of design concepts. The main emphasis of this phase has been to address the technology/advanced development of the aerobrake, avionics/software, cryo-fluid management, cryo auxiliary propulsion, and alternative propulsion systems. This effort is further addressed in the technology/advanced development section of this final report via the roadmaps. The particular schedule driver, as it exists today, is the development of the "smart" aerobrake. Our test program has been established to require the equivalent of an AFE II, whereby the LTS configuration aerobrake (although not full scale) is demonstrated using a "to be" scheduled STS flight in the 1997 timeframe. The development of the smart aerobrake also uses data gathered during the already scheduled AFE I, in the 1995 timeframe.

The STV/LTS phase C/D development testing program has been designed to aid in the preliminary evaluation of the design feasibility and manufacturing process while providing confidence that the flight hardware will pass qualification tests. The test units consist of: 1) structural components of the tank structure, mechanisms etc.,; and 2) component and subsystem breadboards of the RCS, propulsion subsystem, avionics boxes, thermal control subsystem, and power subsystems, etc. To aid in the development of the components and subsystems a functional test unit (FTU) will be required. This unit would be the tool for evaluation of the avionics and control subsystem for software testing. Also the use of engineering prototypes, mockups, and development units would be employed to verify the subsystem design integrity.

The STV/LTS phase C/D qualification test program has been established to verify flight type components, subsystems, and systems meet performance and design requirements under anticipated operational regimes and environments. The test units consist of:

- •Components which would be qualified to M-67-45, Test Methods and Controls, Components
- •A subsystems structural test article (STA) for static load test to demonstrate design integrity of the primary structure and verify the manufacturing process, for Modal Survey to acquire correlation data for Dynamics Model, and for ordnance separation to verify mechanical shock induced by the separation system (i.e.. drop tank separation).
- •A subsystems propulsion test article (PTA) for cryogenic pressure, leak, flow and ignition testing to acquire model correlation data and establish flow induced structure dynamics.
- •Also the subsystems functional test article (FTA) would continue to be used for qualification of the avionics and software.
- •A systems environmental/pathfinder test vehicle (ground test unit - GTU) to be used for thermal vacuum testing to verify capability of meeting operational requirements when subject to vacuum and temperature extremes, thermal extremes, and also to verify that the thermal control subsystem will maintain and control the external subsystems and components to within the design specifications. This unit would also be used for acoustics testing to verify structural integrity of the system for high frequency vibrations, EMC testing to provide data assuring STV/LTS system and STV/LTS launch system compatibility, and pathfinder processing at KSC to validate ground processing and handling procedures.
- •A systems flight test vehicle (FTV) to be used for full-scale flight demonstration of the

STV/LTS unit (with crew module) from ground processing through a full lunar mission prior to the first manned mission.

Figure 5.2-1 defines the subsystem requirements by test article.

Figure 5.2-2 presents the mission objectives accomplished by each flight article.

Figure 5.2-3 displays the test configuration usages.

The timeline in Figure 5.2-4portrays the ground processing, LEO processing, and STV/LTS flight test.

The STV/LTS acceptance and operational test programs would be used to verify flight hardware performance in accordance with design and manufacturing documentation. STV/LTS test units will have an acceptance test performed verifying that the hardware is of known configuration (components, subsystems, and systems). The operational testing would consist of manufacturing in-line acceptance tests, systems operations testing (as practicable on ground and prior to LEO

Subsystems	Structural Test Article (STA)	Propulsion Test Article (PTA)	Functional Test Article (FuTA)	Ground Test Vehicle (GTV)	Flight Test Vehicle (FTV)	Total Subsys Required
Structures/Mechanisms	1			1	1	3
Skirts and Adapters	1			1	1	3
Basic Structure	1			1	1	3
Insulation			1	1	1	3
Heaters			1	1	1	3
Thermal Control			1	1	1	3
Tanks		1		1	1	3
Propulsion System	Mass Sim	1		1	1	3+
Engines	Mass Sim	1		1	1	3+
RCS	Mass Sim		1	1	1	3+
GN&C	Mass Sim		1	1	1	3+
Data Mgt/Sequencing	Mass Sim		1	1	1	3+
Power Gen/Storage	Mass Sim		1	1	1	3+
Telemetry Tracking and Command	Mass Sim		1	1	1	3+
Power Distribution/ Harness	Mass Sim		1	1	1	3+
Environmental Control	1*		1	1	1	4
Aerobrake	1*		1	1	1	4
Crew Module	1*		1	1	1	4

* = Separate test articles from structures STA
+ = Quantity plus mass simulator required

Figure 5.2-1 Subsystem Requirements by Test Article

Test Article / Mission Phase	Aeroassist Flight Experiment I	STV Flight Demo (Aerobrake	Polar Servicing Mission (STV)	Flight Test Vehicle (FTV)	1st Cargo Flight (Flt-0)	2nd Cargo Flight (Flt-1)	1st Piloted Mission (Flt-2)
On-Orbit Assembly and Checkout			√	√	√	√	√
Rendezvous and Docking			√	√	√	√	√
Trans-Lunar Injection (TLI)				√	√	√	√
Descent				√	√	√	√
Ascent				√	EXP	√	√
Trans-Earth Injection (TEI)				√	EXP	√	√
Aeropass Maneuver	√*	√*	√	√	EXP	√	√

* Scaled Configuration Versions of Aerobrake - AFE I Mainly Dynamics Analysis/CFD Modeling, STV Demo Scaled Version of LTV Aerobrake (Rigid or Flexible Still To Be Determined).
Note: FTV To Be Reusable and May Require Refurbishment Prior To Next Usage.
EXP : Denotes That Unit Is An Expendable Unit

Figure 5.2-2 Mission Objectives Accomplished by Flight Article

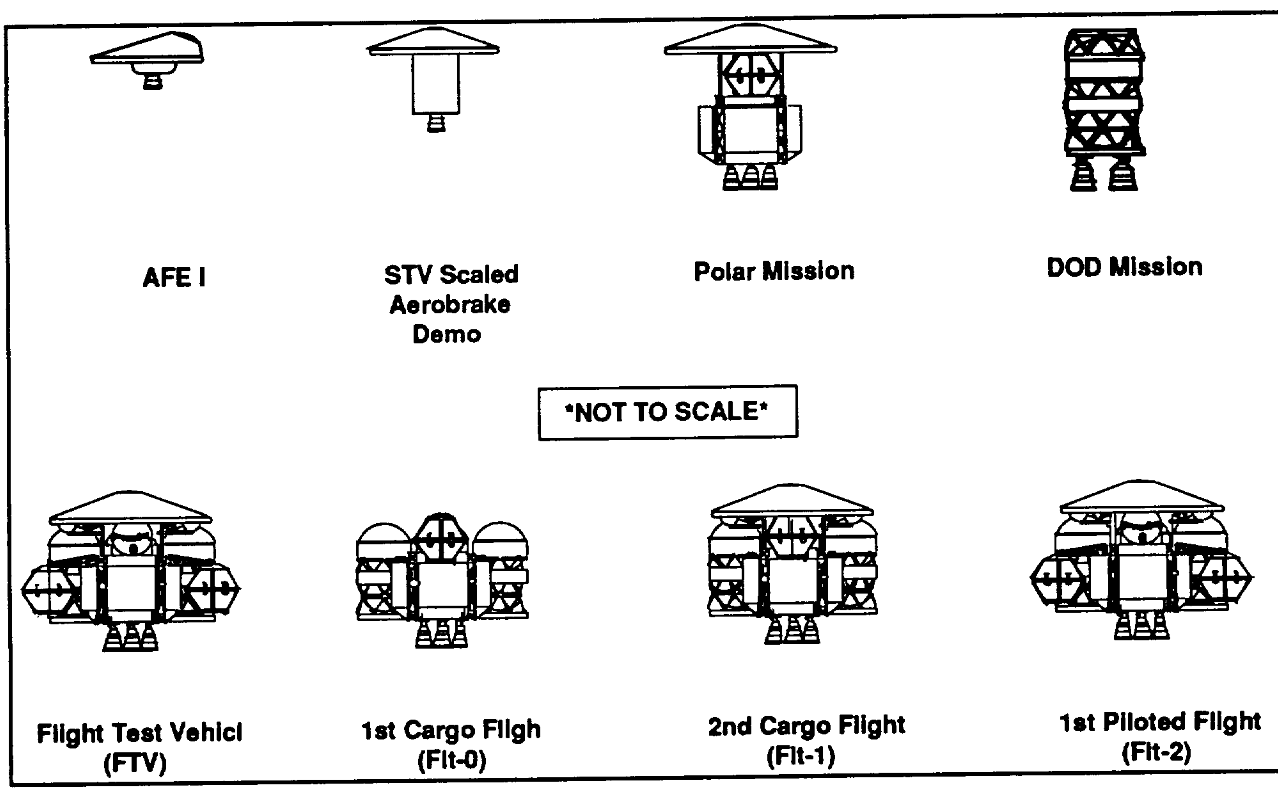

Figure 5.1.4-3 Test Configuration Usages

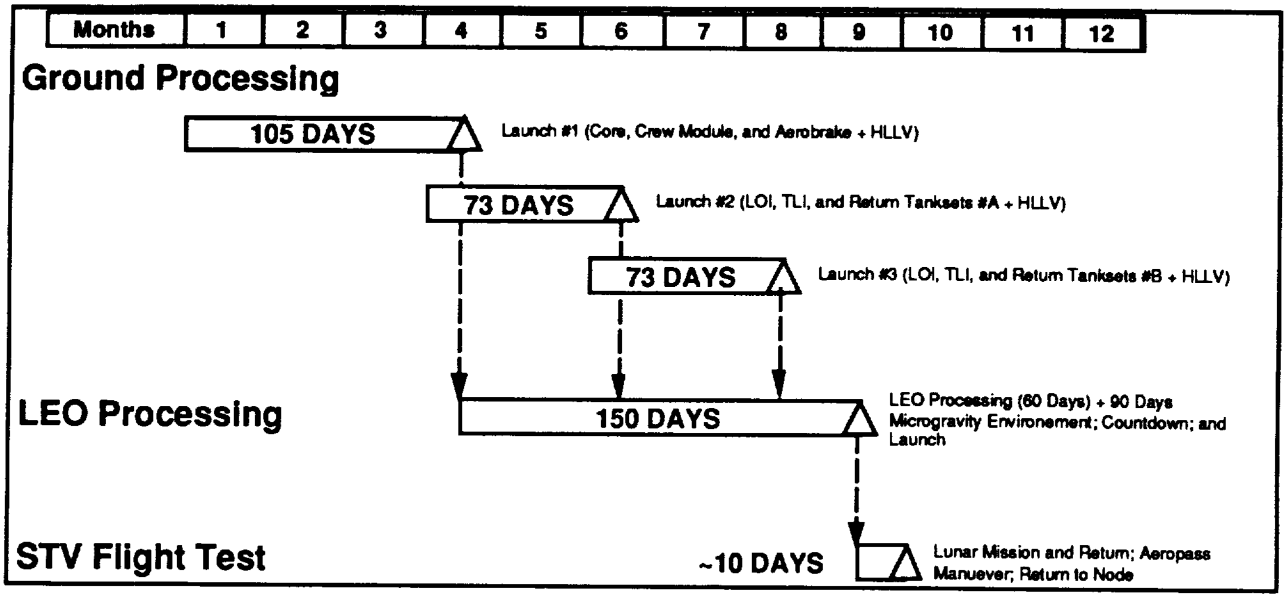

Figure 5.2-4 Ground Processing, LEO Processing, and STV/LTS Flight Test

node departure), and launch processing tests (again as practicable at KSC and prior to LEO node departure). It is expected that much of the testing could and would be accomplished, via built-in-test (BIT) both at KSC and at the LEO node. Launch processing tests would include interface verification, RF verification, STV/LTS system functional, and booster integration and combined system test.

5.3 NETWORK LOGIC MODELS

The network logic model is the beginning of what has become the single database from which we derive the program intermediate and master schedules. Utilizing Open Plan software, the tasks that are to be accomplished have been modeled, interrelationships have been defined, and the anticipated task durations established through assimilation of historical data, lessons learned, and project technical experience. The network logic model has been developed so that all that is required to run the program analysis is that the user load a Phase C/D ATP target date into that activity. The analysis performed will provide the calculated dates for all the activities leading up to and including initial launch capability date for the LTS-TST vehicle, first cargo flight date, first manned mission capability date, and on through the total lunar program as currently manifested. The program has not been constrained to accomplish only one LTS flight per year but is allowed to determine the LEO departure and mission operations spans based upon anticipated processing and launch constraints both on the ground and in LEO. Additionally, the network database is coded by WBS element to the third level, enabling us to sort and select on any single or unique grouping of elements. The network logic model database will be used to drive down program planning requirements to the detailed schedule level and then monitor program activities and status while also serving as a what-if tool for the analysis of potential and real changes to the program baseline. The network logic model flow chart, which is in logic Gantt format, is included in the programmatics appendix.

The program critical path(s) and schedule risk analyses have been accomplished and evaluated. The critical path of the network is the longest path by time through the network or the path through the network with the least float/margin. By analyzing the network it is determined that the primary path flows through engineering design, system PDR, system CDR, aerobrake component and system qual testing, and fabrication of the production units. It must be emphasized that the close margin between the aerobrake, crew module, avionics/software, and engine development programs makes for an extremely difficult task of identifying of the most probable program tall-pole. As the technology and advanced development programs become further defined, the network model can be updated and/or modified to address program changes and critical path analysis reapplied. The scheduled risk analysis was conducted on our network model utilizing the Open Plan Extension for Risk Analysis (OPERA) software package. Activities and key milestones on the critical path were identified and coded with minimum (most optimistic) and maximum (worst case) task durations. Using probability theory and a Monte Carlo simulation, OPERA determines the most likely outcome. The Monte Carlo simulation recalculates the critical path multiple times to account for the relative effort of all possible scenarios. The result is a statistically calculated scenario that predicts the eventual course of the project. The results of the schedule analysis indicated that we have an 87% probability of achieving the LTS-TST flight ILC from the LEO node as scheduled on September 26, 2003.

5.4 NETWORK LOGIC DERIVED MASTER SCHEDULE GANTT CHARTS

The network logic model derived Gantt schedule charts were generated from the network logic database to assist the MSFC Phase C/D STV implementation planning and to identify the time phasing of support programs. The schedules provide a more detailed breakdown of the effort portrayed in the summary master schedules referenced in Section 5.1.3, Summary Master Schedules. The schedule charts are laid out to clearly identify the major program milestones, major subsystem development, major subsystem integration, software development programs, support equipment development and delivery, tooling design and development, data, training, vehicle processing, vehicle launches, and LEO processing, mission operations, systems engineering programs, and support services. The logic and subsequently derived schedules use a "green light" approach to building and testing the subsystems and systems. This implies that the hardware can be built and stored in advance of ground processing and launch to the LEO assembly point. The network logic can be altered to address funding shortfalls by limiting the builds per year to any given set of criteria. The network logic derived master schedule Gantt charts are included in the programmatics volume appendix.

5.5 COST SUMMARY

5.5.1 Top Level Cost Summary

Table 5.5.1-1 shows the STV top level cost by program phase and by major WBS element. It includes the production and launch of 22 vehicles with a LCC of $9809.9 M. The DDT&E cost is $624.4 M, the production cost is $1205.4 M ($55 M average unit cost), and the operations cost is $8417.7.M.

Table 5.5.1-1 also shows the overall cost for the LTS program, including the production of 9 vehicles and launch of 25 missions, is $88,620.4 M. The DDT&E cost is $23,385.4 M, the production cost is $6,375.8 M ($708 M average unit cost), and the Integration and Operations cost is $58,859.2 M.

Table 5.5.1-1 Top Level Cost Summary

Element	DDT&E	Prod	Ops	LCC
Space Transfer Vehicle	451.8	871.9	6090.0	7413.7
Growth and Fee	172.6	333.3	2327.7	2833.6
TOTAL	624.4	1205.2	8417.7	10,247.3
Lunar Transportation System	16,918.7	4612.7	42,583.1	64,114.5
Growth and Fee	6466.7	1763.1	16,276.1	24,505.9
TOTAL	23,385.4	6375.8	58,859.2	88,620.4
STV/LTS TOTAL	24,009.8	7581.0	67,276.9	98,867.7

Costs Reported in Millions of 1991 Dollars

5.5.2 Cost by WBS

Table 5.5.2-1 shows the STV LCC breakout by major WBS element. The total DDT&E cost for the LTS program is projected to be $624.4 M. The total production cost for the STV program is projected to be $1205.2 M. The total operations cost for the STV program is to be $8417.7 M.

Table 5.5.2-2 shows the LTS LCC breakout by major WBS element. The total DDT&E cost for the LTS program is projected to be $23,385.4 M. The total production cost for the LTS program is projected to be $6,375.8 M. The total operations cost for the LTS program is projected to be $58,859.2 M.

Table 5.5.2-1 STV Cost by WBS Element

Element	DDT&E	Prod	Ops	LCC
Vehicle	117.8	689.2	0.0	807.0
Software	50.0	0.0	0.0	50.0
Support Equipment	17.7	0.0	0.0	17.7
System Test	67.1	0.0	0.0	67.1
Facilities	50.0	0.0	0.0	50.0
Operations	13.0	0.0	466.4	479.4
Systems Engineering	95.1	103.4	70.0	268.5
Program Management	41.1	79.3	53.6	174.0
Sub Total	451.8	871.9	590.0	1913.7
ETO Costs	0.0	0.0	5500.0	5500.0
Growth and Fee	172.6	333.3	2327.7	2833.6
TOTAL	624.4	1205.2	8417.7	10,247.3

Costs Reported in Millions of 1991 Dollars

Table 5.5.2-2 LTS Cost by WBS Element

Element	DDT&E	Prod	Ops	LCC
Core Stage/Lander (w/ Crew Cab)	2038.9	2538.7	0.0	4577.6
TLI Tanks	68.8	646.6	0.0	715.4
LOI Tanks	60.8	461.1	0.0	521.9
Software	500.0	0.0	0.0	500.0
Support Equipment	867.4	0.0	0.0	867.4
System Test	2965.0	0.0	0.0	2965.0
Facilities	2550.0	0.0	0.0	2550.0
Operations	295.0	0.0	8108.3	8403.3
Systems Engineering	2398.4	547.0	1216.3	4161.7
Program Management	1174.4	419.3	932.4	2526.1
Sub Total	12918.7	4612.7	10257.0	27788.4
ETO Costs	0.0	0.0	32,326.1	32,326.1
LEO Node Costs	4000.0	0.0	0.0	4000.0
Growth and Fee	6466.7	1763.1	16,276.1	24,505.9
TOTAL	23,385.4	6375.8	58,859.2	88,620.4

Costs Reported in Millions of 1991 Dollars

Figure 5.5.2-1 shows the breakdown of the LTS DDT&E costs in ranked order. Figure 5.5.2-2 shows the breakdown of the LTS DDT&E costs by percentage. The LEO node cost makes up the largest single cost at $4000 M (23.6%), followed by the system test cost ($2965 M, 17.5%), facilities costs ($2550 M, 15.1%), the systems engineering costs ($2398.4 M, 14.2%), and the core vehicle costs ($2038.9 M, 12.8%) . Support equipment, software, operations planning, and site activation make up the remaining costs.

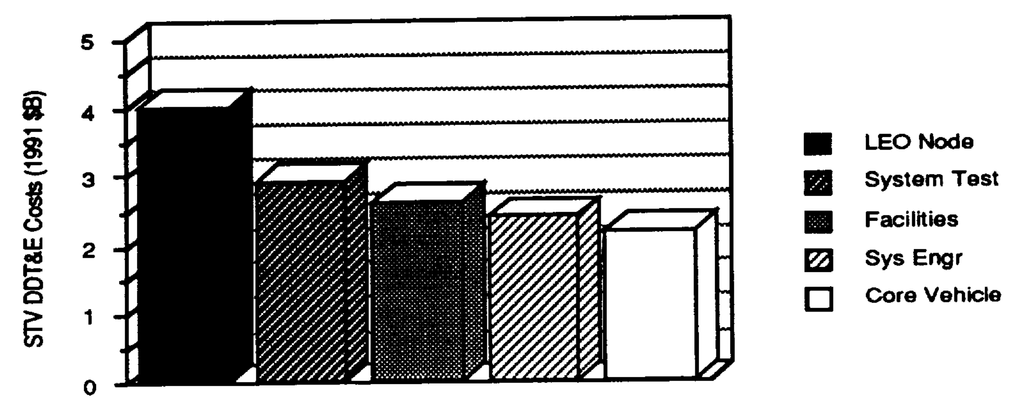

Figure 5.5.2-1 LTS DDT&E Cost

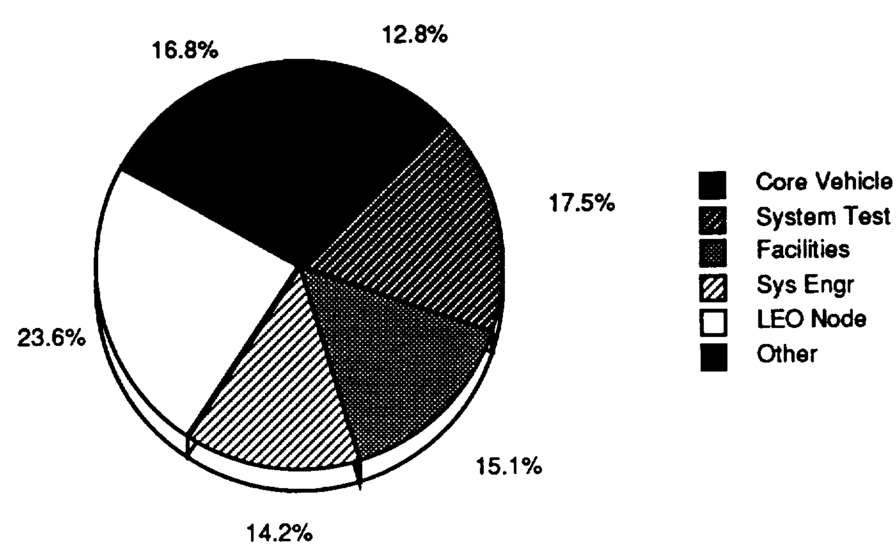

Figure 5.5.2-2 LTS DDT&E Cost

Figures 5.5.2-4 and 5.5.2-4 show the breakdown of the LTS Production costs for 9 vehicles. The core vehicle makes up the largest single cost at $2538.7 M (55.0%), followed by the TLI tank costs ($646.6 M, 14.0%), and the systems engineering costs ($547.0 M, 11.9%). Other costs including the LOI tanks and project management make up the remaining costs.

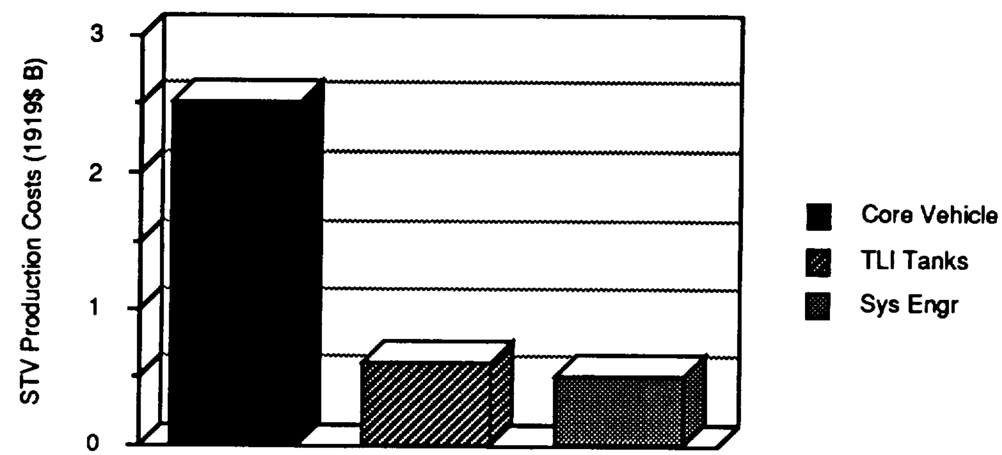

Figure 5.5.2-3 LTS Production Cost

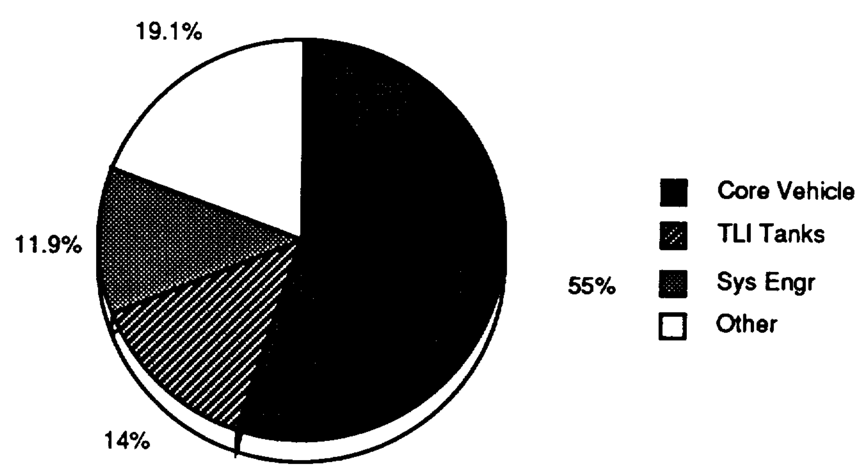

Figure 5.5.2-4 LTS Production Cost

Figures 5.5.2-5 and 5.5.2-6 show the breakdown of the LTS Operations costs for 25 missions. The ETO costs of these missions make up the largest single cost at $32,326.1 M (75.9%), followed by the Operations cost ($8108.3 M, 19.0%). The Systems Engineering and the Program Management make up the remaining costs.

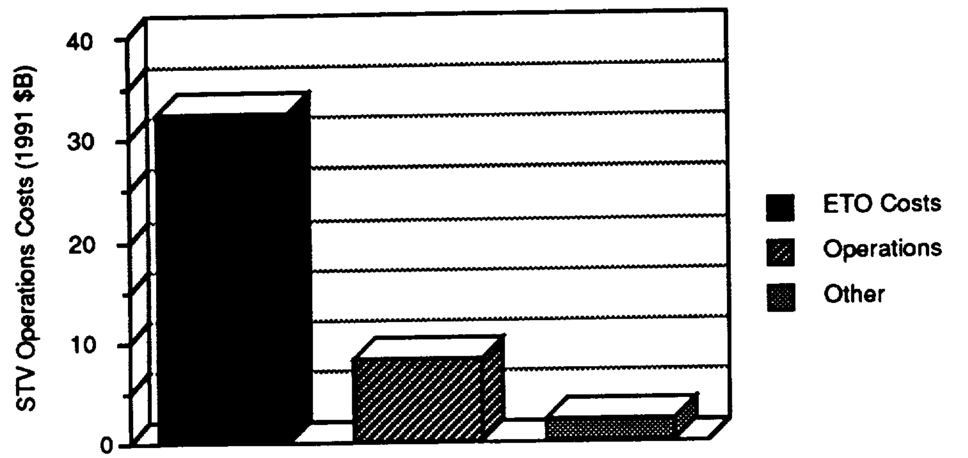

Figure 5.5.2-5 LTS Operations Costs

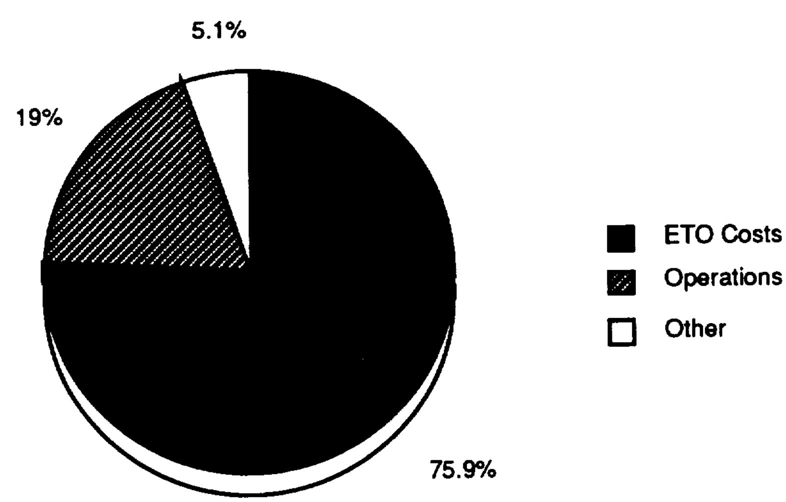

Figure 5.5.2-6 LTS Operations Costs

6.0 TECHNOLOGY/ADVANCED DEVELOPMENT

6.1 Task Objective

The objective of this task was to determine the technologies and advanced development concepts essential for the evolution of the next generation of lunar space transfer vehicles. To satisfy this objective, a comprehensive listing of candidate technologies and advanced development concepts was identified, categorized into nine lunar and three Mars areas, coordinated with MSFC, and screened to eliminate those that have already reached system level maturity. The current maturity level of each remaining technology and advanced development concept was then established, as was a schedule for advancing that level. The focused technology maturation program schedules progressive maturity into the technology and advanced development activities which provide the increasing levels of confidence required by program management for decisively choosing from identified alternative designs or operational concepts. The key to success is flexible, adaptive management of program control, authority and responsibility, with implementation shared by the organizations able to perform the invention, development, demonstration, and implementation with credibility.

6.1 Approach

The STV Technology and Advanced Development (TAD) effort has identified the highest priority technologies and advanced concepts that are essential for the development of lunar STVs which can evolve into vehicles for Mars manned and cargo missions. In order to establish the status of each key TAD concept, development schedules have been defined for each area showing the current TAD maturity level and the existing/planned programs which will advance each TAD concept.

A cost and performance benefits assessment is underway for each candidate TAD concept to quantify its value to the STV program. The process for this effort is shown in Figure 6.1-1. All candidate concepts will be prioritized and detailed development plans will be completed for those with the highest priority. A wide range of technologies have been identified and assessed to ensure the requirements for all STV concepts being evaluated are considered. All TAD concepts will be prioritized based upon their impact on STV cost, performance/safety and development schedule. Those that have a significant effect on any of these three criteria will be identified as "High" priority items. Those that have a moderate effect will be identified as "Medium" priority, and a "Low" priority will be assigned to those which have an insignificant effect on STV cost,

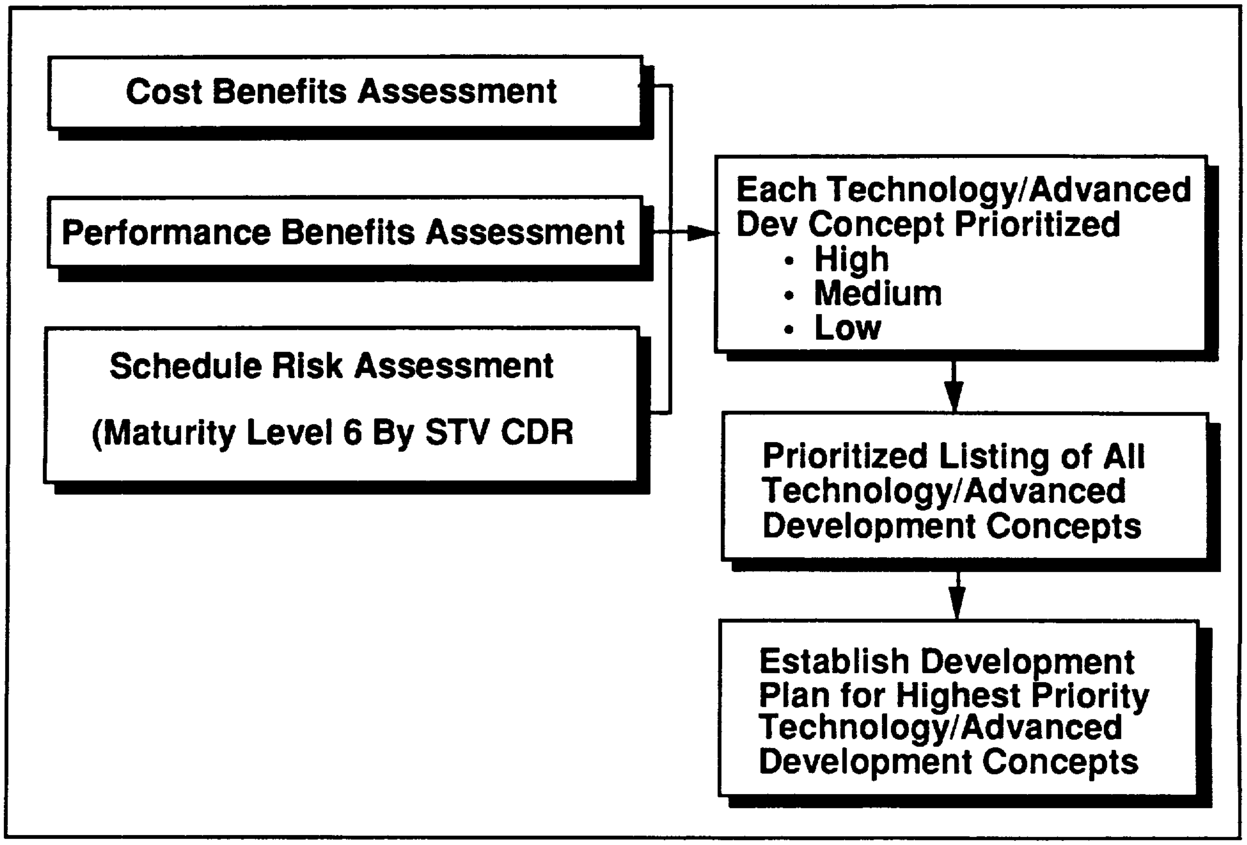

Figure 6.1-1 Technology/Advanced Development Analyses Process.

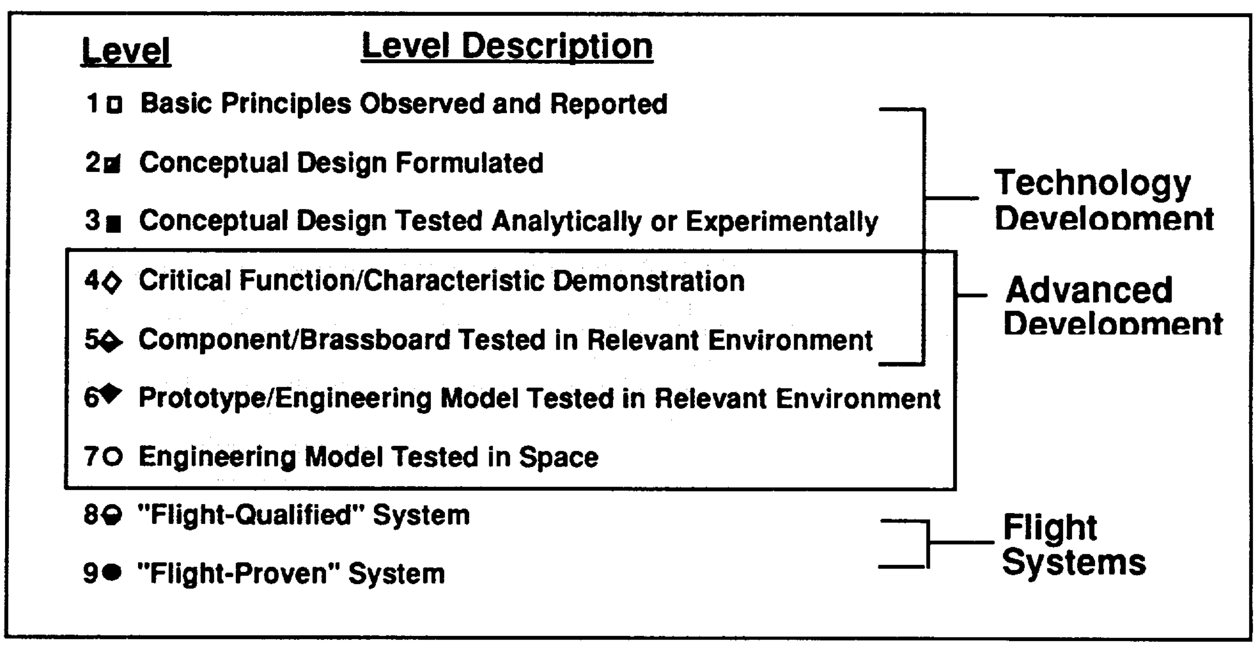

Figure 6.1-2 TAD Maturity Level Definitions

performance or schedule. All the TAD concepts evaluated in this study will be listed according to their priority and a development plan will be established for the highest priority concepts.

TAD Maturity Levels—Definitions of the seven TAD maturity levels illustrated in Figure 6.1-2 were derived from the NASA Space Systems Technology Model (January 1984). They range from the observation of the basic principles (Level 1) to an engineering model tested in space (Level 7). To minimize program risk with resultant cost overruns, it is imperative that a maturity Level 4 be reached by STV Preliminary Design Review and a maturity level of 6 (with 7 preferred) be obtained by the Critical Design Review (CDR), tentatively shown as the first quarter of 1997.

The twelve basic, top-level STV system requirements that drive the technology and advanced development needs are summarized in Table 6.1-1. Although the first five listed have slightly more impact on almost all the major STV systems than the other seven, all twelve directly affect the selection of the key technology and advanced development concepts.

Table 6.1-1 STV Requirements That Drive Technology/Advanced Development

• Evolve For Mars Missions
• Manrated, Dual Fault Tolerant & High Reliability
• Withstand Space Environments, Long Duration
• Robust Design, Margins
• Minimum Space Assembly & EVA and No In-Flight Maintenance
• Cryogenic Propellant, 5 to 12 Months Propellant Storage
• In-Space Fluid Management & Transfer *
• Minimum In-Space Fluids
• Aeroassist GEO, LEO or Mars Return *
• Autonomous Rendezvous, Docking & Landing *
• In Situ Resources
• Low Life Cycle Costs and Acceptable Performance
• If Hardware Reused, 5 to 30 Year Service Life
* Not Required For All Concepts

6.2 Key STV Areas

Table 6.2-1 shows the ten key STV technology and advanced development areas essential for the development of lunar STVs that evolve into Mars vehicles. Early GEO vehicles will incorporate less advanced technology/development concepts and serve as test beds for the more advanced concepts required for sustained lunar, Mars and planetary travel.

In-depth development schedules have been prepared for each of the twelve TAD areas. These schedules show the current maturity level, the on-going programs (if any) that will be raising the maturity level, and the agency or program that is responsible for increasing the maturity. Only a portion of one schedule is shown here due to space limitations. Schedules for all TAD concepts are available upon request.

6.2.1 Aerobrake

The aerobrake has significant cost and performance benefits compared to an all propulsive stage, thus, TAD concepts critical to the design, fabrication and control of aerobrakes were identified as high priority

Table 6.2-1 Key STV Technology/Advanced Development Areas.

Area	GEO	Lunar	Mars
• Aerobraking	√	√	√
• Avionics	√	√	√
• Cryo Fluid Mgmt	√	√	√
• Cryo Space Engine	√	√	√
• Space & Ground Operations (Robotics, AI, etc.)	√	√	√
• Crew Module		√	√
• ECLSS		√	√
• Cryo Auxiliary Propulsion		√	√
• Alternative Propulsion		√	√
• In Situ Resources		√	√

items. As can be seen in the aerobrake TAD development schedule (Fig. 6.2-1), several will reach level 5 to 6 maturity as a result of the aerobrake flight experiment (AFE) which is to fly in 1994. However, except for atmosphere characterization, there are no current programs identified which will advance any of the TAD concepts past this level. In the case of fault tolerance and space environmental effects, there are no programs identified to date that would advance them past their current maturity levels of 2.

Additional TAD concepts related to the aerobrake development (not shown in Fig. 6.2-1) include TPS materials and structures (advanced rigid tiles, carbon-carbon materials, intermetallics, flexible blankets, low density ablators, and high temperature structures), health and status monitoring, and testing of high temperature materials. There are no current programs which will develop any of these concepts to a level 7 except for high temperature structures. Development of carbon-carbon by the Survivable Solar Panel Power System (SUPER) and NASP programs may help advance this material, but not to the extent required for the aerobrake thermal protection system. Neither aerobrake health and status monitoring nor high temperature materials testing have any programs identified to date which will advance them past their current maturity level of 2.

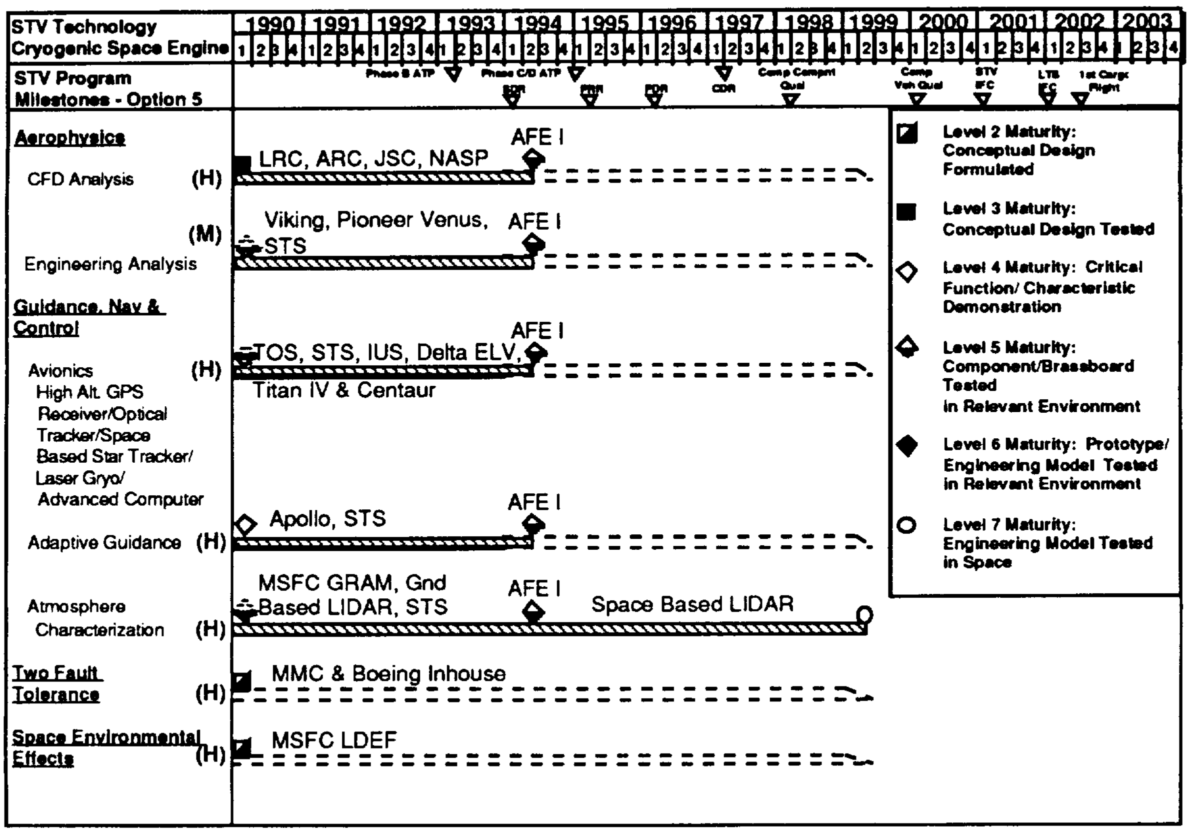

Figure 6.2-1 Aerobrake TAD Schedule.

6.2.2 Avionics

TAD concepts applicable to the STV avionics subsystem include health and status monitoring (architecture, two fault tolerance, redundancy management, and synchronization), computers (architectures and advanced memories), software (multi-redundancy in real-time operation, computer generated codes, and automatic verification and validation), bus architectures based on fiber optics, photonics, power (fault tolerant distribution, high density batteries, and advanced fuel cells utilizing fuel grade cryos), communications (Ka band, advanced S-band, laser array antenna), analytical models, space environmental effects including SEU and development of a national avionics test bed. Only one of these concepts, fault tolerant power distribution, is expected to reach level 7 maturity through a currently funded program. Only one other avionics TAD concept, communication, has any on-going development programs. None of the remaining avionics TAD concepts are anticipated to advance beyond level 5 maturity through any current or projected programs and several are not expected to matured beyond level 3. Six of the avionics TAD concepts are ranked as high priority in the initial prioritization for STV application, and three ranked as medium priority.

Table 6.2.3-1 Cryogenic Space Engine TAD Concepts.

(H)	**Throttling** - 10:1 to 20:1 - Tank Head & Pumped Idle Modes - Zero NPSP - Efficient Injectors & Combustion Stability
(M)	**Advanced/High Speed Turbopumps**
(M)	**Efficient High Pressure Combustion Chamber**
(M)	**Large Extendible/ Retractable Nozzles and Deployment Mechanisms**
(M)	**Electromechanical TVC Actuator**
(M)	**Electromechanical Control Valves**
(H)	**Integrated Modular Engine**
(H)	**Health & Status Monitoring**
(H)	**Cryogenic Engine Test Facility**
(M)	**Analytical Models**
(H)	**Two Fault Tolerance**
(H)	**Space Environmental Effects**

6.2.3 Cryogenic Space Engine

Table 6.2.3-1 lists twelve TAD concepts related to development of cryogenic space engines, along with our preliminary assessment of the priority to the STV program. Six of the twelve TAD concepts identified as critical to the development of STV cryogenic space engines have been assessed to be high priority and none will have advanced past Level 5 maturity under current programs at time of STV CDR, and some will only be at maturity Level 4. Only one of the medium priority TAD concepts will have reached Level 7 maturity through existing programs and, again, this will occur past the STV program CDR.

6.2.4 Cryogenic Fluid Management

Table 6.2.4-1 CFM TAD Concepts

Storage
(M) AL-Li Tanks
(M) Composite Tanks
(H) Robust Insulation
(H) Refrigeration
(H) Reliquefaction
(M) VCS
(M) Mixer Pump
(M) TVS

(H) **Space Environmental Effects**

(H) **Health & Status Monitoring**

(H) **Two Fault Tolerance**

Instrumentation
(M) Low g Mass Gauging
(M) Smart Temperature Sensors
(M) m Smart Pressure Sensors

Transfer
(M) Automated Propellant Loading
(H) Cryo Disconnects/Couplers
(M) Composite Cryo Transfer Line
(M) No-Vent Fill

Acquisition
(L) Engine Feed Start Basket
(M) Liquid Acquisition Device

(M) **Electromechanical Vent Valve**

(M) **Slush Transfer, Storage & Gaging**

(M) **Analytical Models**

The TAD concepts for the STV CFM area have also been identified, development schedules defined, and an assessment made of all the existing and planned programs that would develop each TAD concept through a maturity level of seven. All but two concepts shown in Table 6.2.4-1 are assessed to have a medium or high priority. Fourteen will require new or expanded programs to bring them to level seven maturity. Four concepts have current or planned programs to bring them to level seven maturity and are considered acceptable STV program risks because they will have reached level four maturity prior to the PDR and level six prior to CDR. The 10 TAD areas shown are listed in order of importance to the STV program. Cryogenic storage was ranked the highest because propellant boiloff affects crew safety (adequate propellant to perform the mission and return), performance (tank size and weight), and cost (mass placed in LEO). Development of robust insulation concepts has also lagged behind most other CFM technologies/advanced developments. It will, therefore, have an adverse effect on STV development and schedule risks unless it is adequately addressed in time to support STV missions which require cryogenic propellant storage greater than a few hours. Space environmental effects were ranked second because there is little specific information on the long term effects of space (atomic oxygen, UV, space debris and meteorites, etc.) on CFM hardware. Health and status monitoring and two fault tolerance were also considered to have high priorities because both are essential to the safe operation of STVs and both will have significant impacts on STV performance and cost. The remaining TAD concepts were ranked in the order shown based upon their potential effect on STV development, cost and performance.

6.2.5 Cryogenic Auxiliary Propulsion

The TAD concepts related to cryo auxiliary propulsion include turbopumps, liquid acquisition devices, heat exchangers, thrusters (both 25 lb fixed and 50-1000 lb throttleable), and integrated RCS. The concepts presently being pursued which would support the development of gaseous hydrogen/oxygen reaction control systems (RCS) are the liquid acquisition device and throttleable modular engines. Although extensive testing has been performed on 25 lb RCS thrusters for the space station, current analyses show that much larger thrust systems (50 to 1000 lb thrust) are also required for the candidate STV vehicles. A throttleable RCS from 50 to 1000 lb thrust would have significant cost and schedule savings and, thus, rank high on the TAD priority list. Although this system would benefit from on-going NASP research, additional development may be required to bring it up to Level 7 maturity.

6.2.6 Alternative Propulsion

Various nuclear thermal rocket and electric propulsion concepts could be used for lunar mission, but are better suited for the longer duration Mars trip. Because both of these propulsion concepts significantly increase payload capability and/or reduce trip times, the associated TAD concepts have been identified as high priority for Mars missions only. There are virtually no current programs which will advance the maturity level of the nuclear propulsion TAD concepts. Several programs presently under way to develop the various electrical propulsion TAD concepts may be adequate to support the Mars exploration missions, as the Mars vehicles are not required until well after the year 2000.

6.3 Cost and Performance Benefits Analyses

To quantify the cost and performance benefits of each TAD concept, an analysis is being performed using the Zero Base Technology Concept (ZBTC) approach developed on the Advanced Launch System (ALS) program. In this approach, a reference ZBTC is defined and its Life Cycle Cost (LCC) and performance established. The cost and performance effects each TAD concept has on the ZBTC is then assessed. For our analysis, the Martin Marietta 90 Day Study vehicle reference concept was selected as the ZBTC. This reference vehicle was assumed to use existing technology and hardware such as RL-10A-4 engines, aluminum tanks and aluminum-mylar MLI. The non-recurring, recurring, and LCC for the ZBTC are shown in Figure 6.3-1. This analysis assumes five flights per vehicle.

A detailed breakdown of the ZBTC reference vehicle recurring cost shows the largest item for the LTV is the crew module, which is closely followed by the tanks and subsystems costs. Other significant cost contributors are the structure and propulsion systems. For the LEV, the crew module again makes up the greatest part of the recurring cost, with the structure being a distant second. The propulsion system accounts for only about 12% of the total LEV recurring cost and tanks less than 8%.

When the cost and performance benefits analyses have been completed for each candidate TAD concept, they will be ranked against each other based upon the total LCC savings. To ensure that each concept is assessed properly, data will also be derived as to the concept's total investment cost, recurring savings per flight, cost benefit (LCC divided by research and technology cost), and net present value for a 5% discount rate. All this information will be used to establish the "cost"

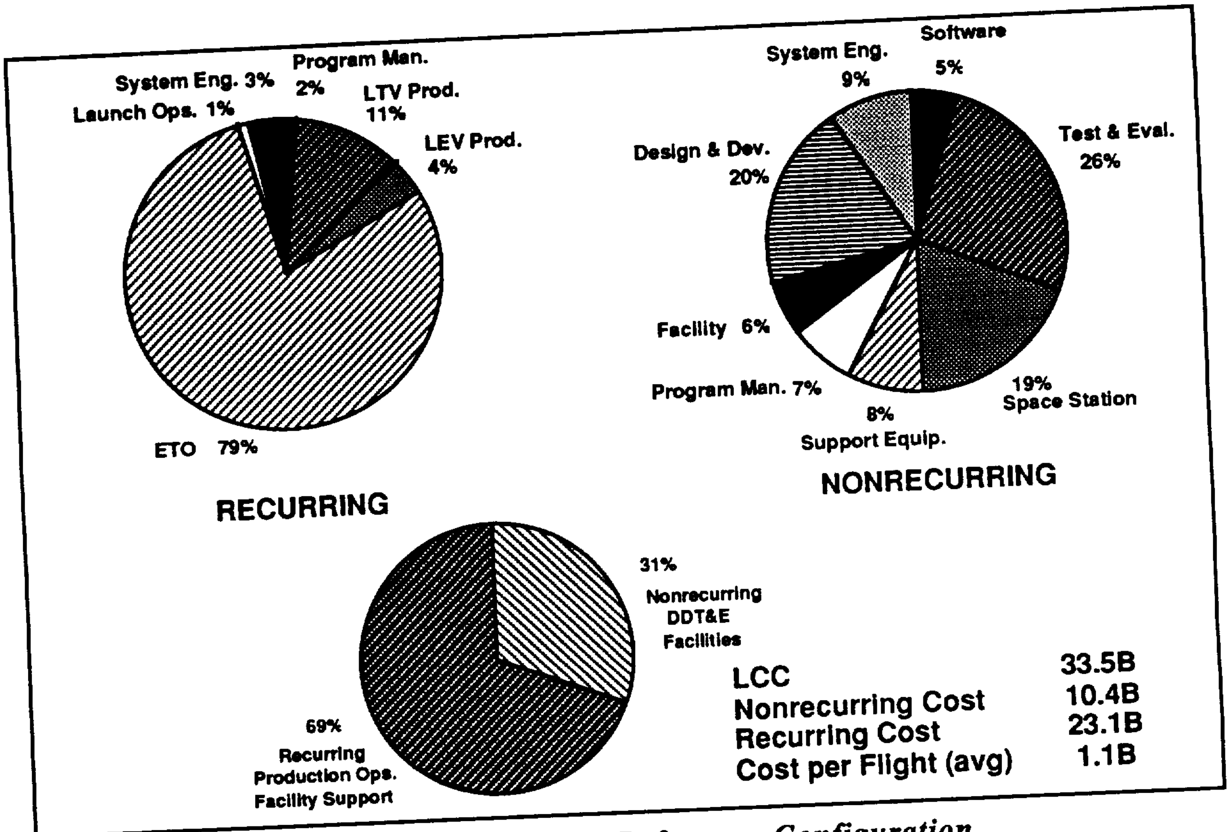

Figure 6.3-1 LCC of ZBTC: 90 Day Reference Configuration.

ranking which will be integrated with the "performance" and "schedule" rankings to arrive at the high, medium and low priorities for all of the STV TAD concepts.

Results from the initial assessment of the TAD concepts show the potential high priority items to be aerobrake aerophysics, guidance/control and materials; avionics, power, software and fault tolerance system; cryogenic engine throttling and integrated modular engine; health and status monitoring; fault tolerance and space environmental effects. Our study results show that many of the potentially high and medium priority TAD concepts will not reach an adequate level of maturity to support the STV program without additional funding.